SID 67-373-1

FINAL REPORT

A STUDY OF ELECTRONIC PACKAGES ENVIRONMENTAL CONTROL SYSTEMS AND VEHICLE THERMAL SYSTEMS INTEGRATION

SUMMARY

(Contract NAS8-20320)

21 July 1967

Prepared by

D.J. Watanabe
Project Engineer

Approved by

L. Isenberg
Project Manager

J.A. Stevenson, Manager
Environmental Control and
Life Support Systems

NORTH AMERICAN AVIATION, INC.
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FOREWORD

This report was prepared by the Space Division of North American Aviation, Inc., under Contract NAS8-20320, "Electronic Packages Environmental Control Systems and Vehicle Thermal Systems Integration," for NASA'S George C. Marshall Space Flight Center (MSFC). The work was administered under technical direction of the Propulsion and Vehicle Engineering Laboratory, with F. Huneidi the contracting officer's representative.

This three-volume report presents the results of the 12-month study to determine the optimum environmental control systems for thermally conditioning individual electronic packages for space missions of durations ranging from 4-1/2 hours to 180 days.

The first volume (SID 67-373-1) is the summary, which contains the significant results and supporting information of the technical study.

The second volume (SID 67-373-2) is the technical report, which contains the details of the study effort and the results and conclusions drawn from the study.

The third volume (SID 67-373-3) is the appendix, which contains the CRT data plots of the thermal analysis (heat balances) conducted during the study.

The technical personnel who contributed to the study effort are C.A. Aldrich, R.C. Coda, J. Hermann, B. Mykytyn, and D.J. Watanabe of the Space Division, and H. Kamei of the Autonetics Division, North American Aviation, Inc.

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NOMENCLATURE

TERMS

Apsis	In an orbit, the point at which the distance of the body from the center of attraction is greatest (higher apsis) or least (lower apsis), $\frac{dr}{dt} = 0$
Eccentricity	The ratio of the radius vector through a point on a conic to the distance from the point to the directrix
Epoch	An instant of time or a date selected as a point of reference
Geocentric	Related to or measured from the earth's center; relating to the earth as a center
Heliocentric	Referred to the center of the sun as the origin
Inclination	Angle between orbit plane and reference plane
Perigee	The point on a geocentric orbit nearest the earth's center
Right ascension	The distance eastward or counterclockwise along the celestial equator, from the first point of Aries to the meridian passing through any celestial body
Semi-major axis	The distance from the center of an ellipse to an apsis; one-half the longest diameter; one of the orbital elements
True anomaly	The angle at the focus between the line of apsides and the radius vector measured from the perifocus in the direction of motion

ABBREVIATIONS

AAP	Apollo Applications Program
AORL	Apollo orbiting research laboratory

ASM/IU	Airlock service module/instrument unit
BSM	Basic subsystem module
CRT	Cathode ray tube
CPC	Ceramic printed circuit
CM	Command module
CSM	Command and service module
deg	Degree
ECS	Environmental control system
F	Fahrenheit
FET	Field effect transistor
GN_2	Gaseous nitrogen
GSE	Ground support equipment
HX	Heat exchanger
IC's	Integrated circuits
IMU	Inertial measuring unit
IR	Infrared
IU	Instrument unit
kwe	Kilowatts electrical
kwt	Kilowatts thermal
LEM	Lunar excursion module
LH_2	Liquid hydrogen
LM	Lunar module
LORL	Large orbiting research laboratory

LOX	Liquid oxygen
LRU	Line replaceable unit
MM	Mission module
MMH	Monomethyl hydrazine
MORL	Manned orbiting research laboratory
MOS	Metal oxide semiconductor
n. mi.	Nautical mile
OLS	Orbital launch stage
psig	Pounds per square inch gauge
RTG	Radioisotopic thermoelectric generator
RF	Radio frequency
SCR	Silicon controller rectifier
SLA	Spacecraft/lunar module adapter
SOS	Silicon on sapphire
SSGS	Standardized space guidance system
scfm	Standard cubic feet per minute
TWT	Traveling wave tube
UDMH	Unsymmetrical dimethyl hydrazine
UHF	Ultra high frequency
Veh CP	Vehicle coldplate
vm	Velocity meters

SYMBOLS

A	Heat transfer area, square feet
A_e	Total absorbed energy on a surface, Btu/(hour)(square feet)
A_r	Radiator area, square feet
C_p	Specific heat, Btu/(hour)(F)
D_B	Bolt diameter, inches
D_E	True contact diameter, inches
E	Earth emission, Btu/(hour)(square feet)
E	Voltage across one module
EI	Electrical input power, watts
h_B	Boiling heat transfer coefficient, Btu/(hour)(square feet)(F)
H_i	Initial enthalpy, Btu per pound
K or K $(\overline{T})$	Thermal conductance parameter
h_v	Heat of vaporization, Btu per pound
k	Thermal conductivity, Btu/(hour)(square feet)(F/foot)
l	Heat conduction path length, feet
N	Number of modules
P_c	Fluid critical pressure, pound per square inch absolute
Q	Heat load, watts or kilowatts
q	Heat transfer rate, Btu per hour
q_e	Earth-emitted radiation, Btu/(hour)(square feet)
Q_{el}	Total heat load, electrical equipment, kilowatts
q_{el}	Electrical equipment heat load per coldplate, watts

q_r	Earth-reflected solar radiation, Btu/(hour)(square feet)
R	Reflected solar energy, Btu/(hour)(square feet)
R or $R(\overline{T})$	Electrical resistance parameter
q_s	Direct solar radiation, Btu/(hour)(square feet)
R_f	Heat dissipator equivalent thermal resistance, C/watt
S	Solar radiation, Btu/(hour)(square feet)
T_c	Cold junction temperature, F
t_c	Coolant coldplate inlet temperature, F
T_h	Hot junction temperature, F
T_R	Radiator surface temperature, R
T_S	Effective sink temperature, R
T_s	Coolant supply temperature, F
t_s	Saturation temperature, F
$\overline{T}$	Average temperature, F
T%	Percent of maximum tightening torque
ΔT	Temperature difference
$\dot{w}$ or W_C	Coolant flow rate, pound per hour
W_{WB}	Weight of water boiler, pound
YS_S	Yield strength of material at junction surface interface, psi
YS_R	Yield strength of bolt, psi
α or $\alpha\ (\overline{T})$	Seebeck parameter
α_s	Solar absorptivity
ϵ	Infrared emissivity

ϵ_D	Dissipator effectiveness
ρ_v	Density of saturated vapor, pound per cubic foot
σ	Stefan-Boltzmann constant (0.1713×10^{-8} Btu/hr-ft^2-(R)^4)

INTRODUCTION

With NASA planning underway for use of the Saturn instrument unit (IU) over greater periods of time than that of the present 6-1/2 hour LOR mission and for extended operational period's in future space missions, an important consideration is the control system necessary for thermal conditioning of the astrionic equipment located within the IU. Future Saturn V missions with a variety of mission profiles, vehicle configurations, and mission durations can be expected to impose a wide range of increasingly stringent requirements on the thermal conditioning system (TCS).

This study was undertaken to determine the optimum thermal conditioning systems for IU electronic packages for Saturn space missions ranging from 4-1/2 hours to 180 days. These systems are to be basic concepts and are to be optimized on the basis of mission duration, operational temperature limits, and heat dissipation rate of the electronic packages.

The current IU thermal conditioning system is designed for about six and one-half hours of operation, but with minor modifications and/or additions it is possible that this time could be extended. The replacement of those components with limited disign life and an increase in the amount of stored supplies (water and gaseous nitrogen) may be sufficient in some cases, particularly for relatively short durations. For longer operation, the modification could become more extensive since the weight and volume penalty of stored supplies could become prohibitive. The extent of the necessary modifications and/or additions will depend upon the mission and the on-board astrionic equipment.

With the possibility of extending the current system's operation, and with the high development cost of new systems, the study was directed toward uprating the current IU thermal conditioning system. Important considerations include maximum use of existing components, impact of possible design improvements of astrionic equipment, and changing environmental conditions and heat loads. The recommended system concepts represent a logical evolution from the current thermal conditioning system concept with an increasing degree of modifications. In addition to the system concepts, system data in parametric form that were used in establishing the recommended concepts have been provided.

The study involved a number of tasks that included (1) defining the requirements and constraints imposed by the various missions and vehicle configurations and by the thermal limits of the astrionic equipment;

(2) reviewing the current IU astrionic equipment to establish possible design improvements based on present technology and on projected new developments and resulting impact on thermal requirements; and (3) synthesizing candidate thermal conditioning systems (starting with the current system concepts), conducting system analyses to provide necessary data for systems selection, and selecting the optimum thermal conditioning system concepts suitable for future Saturn missions. Finally, as a result of the study, several conclusions have been made relative to the recommended system concepts and recommendations for future investigations.

This volume summarizes the results of the 12-month study.

1.0 REQUIREMENTS AND CONSTRAINTS

System requirements and constraints have been established to provide the basic guidelines for the technical effort. The requirements and constraints encompass the range of values or conditions representative for foreseeable future missions. Astrionic equipment thermal requirements and environmental thermal conditions imposed by the mission/vehicle combination provide the basic TCS requirements.

MISSION VEHICLE DATA

For this study, three mission models for the Saturn V have been selected: near-earth orbit, synchronous earth orbit, and lunar and planetary missions. Typical mission profiles have been established. These profiles, in conjunction with vehicle configurations, were used to define the thermal environment and mission/vehicle constraints that may be expected for future missions with 4-1/2 hour to 180-day durations. For the mission models, the assumed mission profiles given in Figure 1-1, together with the assumed vehicle orientations, establish the environmental heat loads expected. The mission phases and duration indicated in Figure 1-1 were assumed to be typical and, thus, adequate for this study.

Several vehicle configurations have been assumed that are considered to be representative of the types that may be selected for the near-term and future Saturn missions. These configurations are basically the Saturn V with various combinations of the IU or units with uprated upper stages and spacecrafts. The current IU/S-IVB configuration was selected as the baseline.

The potential mission vehicle configurations are illustrated in Figure 1-2, which is intended as an aid in defining the overall thermal environment imposed by the vehicle and the space environment. Various potential configurations have been considered to provide the widest possible range of conditions that may be expected for the future missions.

SPACE ENVIRONMENT

For purposes of this study, a relatively simple model for the space environment was assumed. The significant environmental factors are low density or vacuum condition, zero gravity, and thermal radiation. The other environmental factors were considered to have negligible effects, based on the assumption that mission trajectories and operational time periods will be selected to minimize environmental hazards such as radiation and particles.

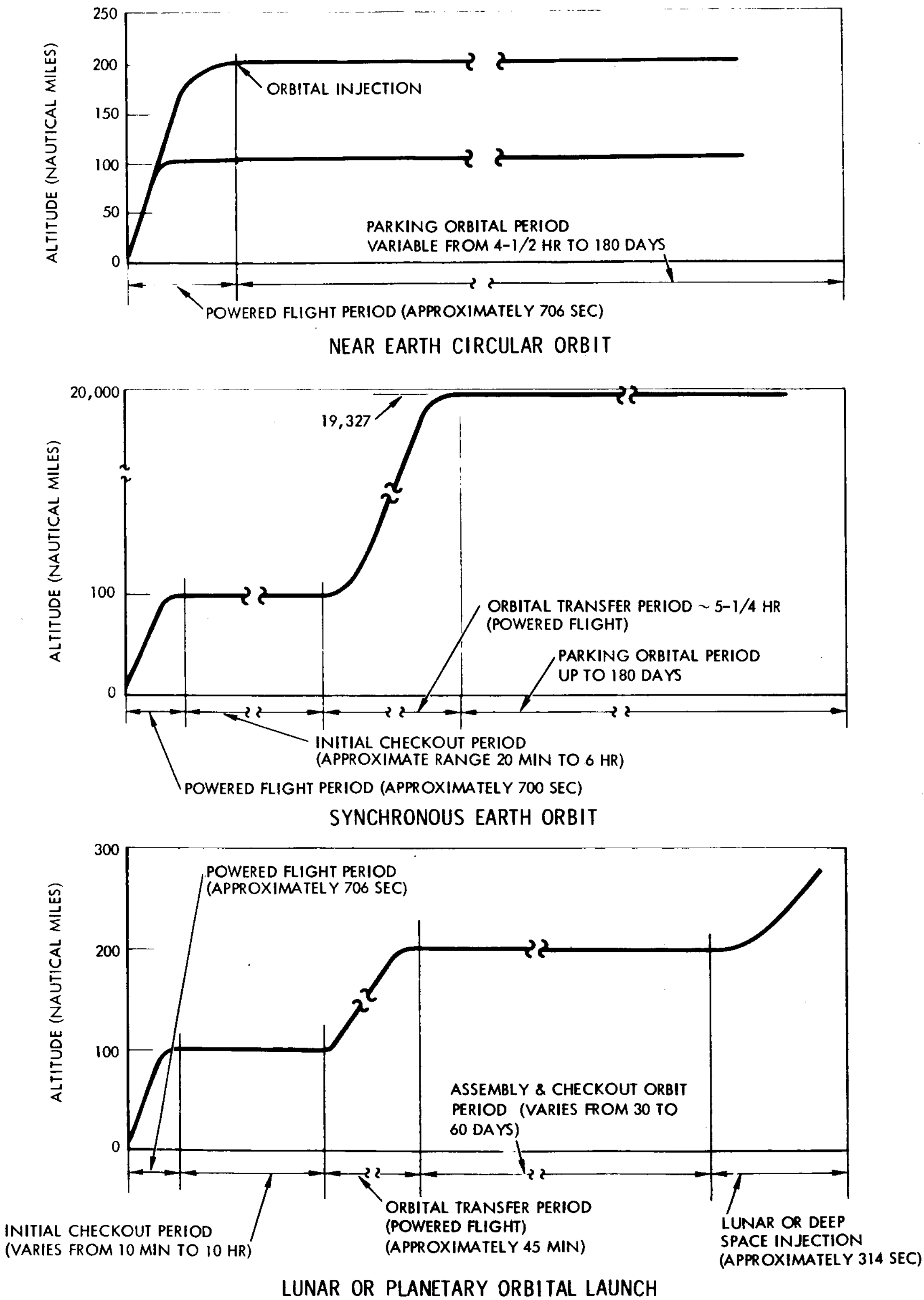


Figure 1-1. Basic Mission Profiles

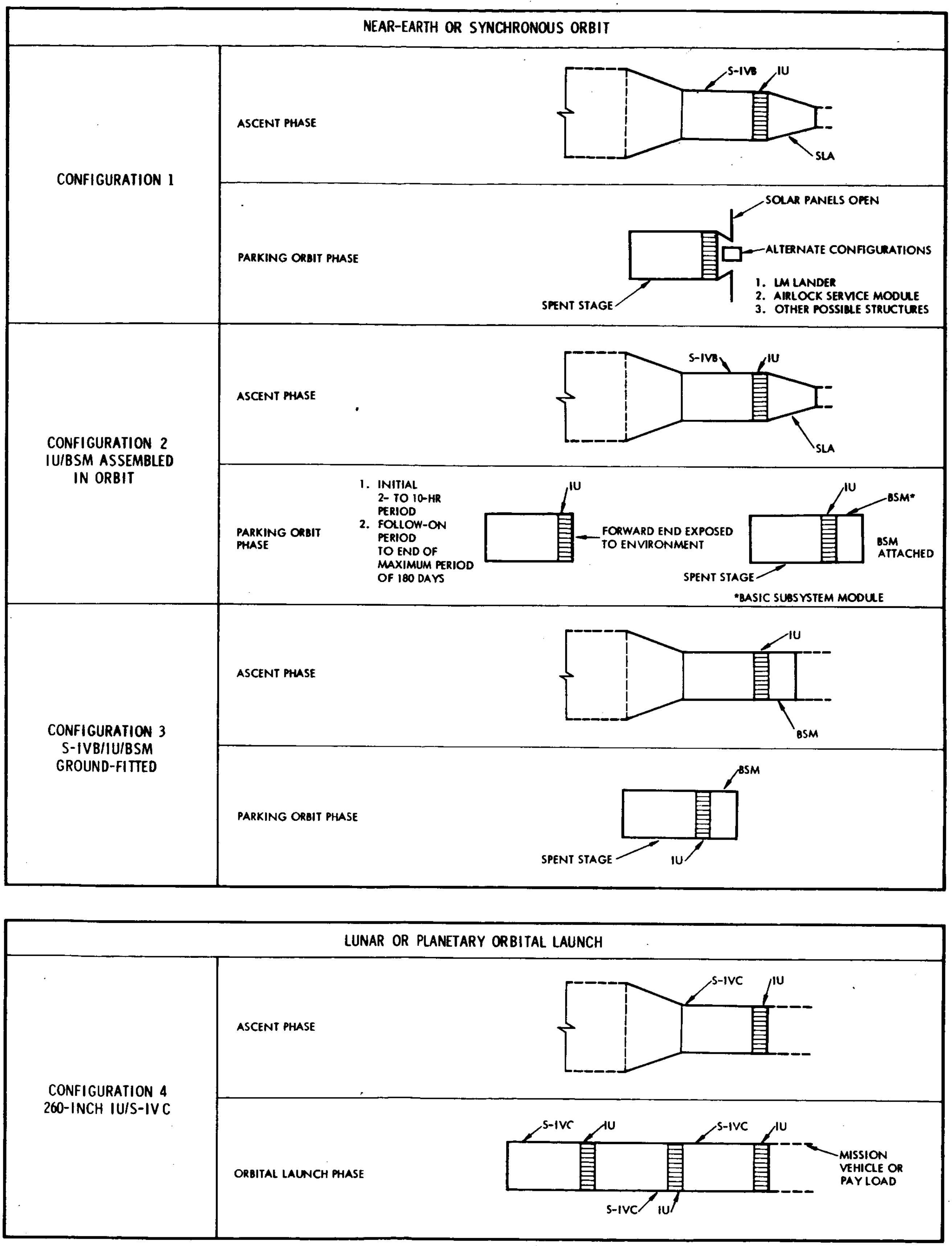


Figure 1-2. Potential Mission Vehicle Configurations

INCIDENT HEAT FLUX

Thermal radiation from planetary bodies striking on the vehicle surface is a prime consideration in the design of the thermal conditioning system. For this reason, incident heat fluxes have been obtained for a simulated spacecraft (or IU) for various orbits and vehicle orientations associated with the three mission models selected for the study. The various orbits and vehicle orientations for which the incident heat fluxes were computed are identified in Table 1-1. In this table, Cases IU-1, IU-2, IU-3, and IU-11 describe orbits and vehicle orientations applicable to the mission profile for a 200-nautical-mile near-earth circular orbit, as shown in the upper portion of Figure 1-1. Cases IU-1, IU-2, and IU-11 are also applicable to the assembly and checkout orbit period for a lunar or planetary mission (see lower portion of Figure 1-1). The 100-nautical-mile near-earth circular orbit and the initial checkout orbit period for a lunar or planetary mission are described by Cases IU-7, IU-8, and IU-10; and the initial checkout orbit period for the synchronous earth orbit mission is described by Case IU-4. The synchronous orbit itself is considered in Cases IU-5 and IU-6, and Case IU-9 applies to lunar or deep space injection.

The relationship of the vehicle coordinate axes, IU location numbers, and vehicle orientation is illustrated in Figure 1-3 for a 200-nautical-mile earth orbit at 29-degrees inclination. The positive y axis was chosen to pass through IU Location 21, the positive z axis was chosen to pass through IU Location 15, and the positive x axis was chosen to coincide with the vehicle longitudinal axis. The three vehicle orientations used in the calculation of incident heat flux are illustrated for the position of the vehicle at the sub-solar point. When the x axis is tangent to the flight path (x-tan), the negative z axis (IU Location 3) faces the earth continuously. When the x axis is sun-oriented (x-solar), the forward end of the vehicle faces the sun continuously. When the y axis is sun-oriented (y-solar), the positive y axis (IU Location 21) faces the sun continuously.

These incident heat fluxes were obtained by using an IBM 7094 digital computer program developed at NAA that calculates direct solar radiation, planet-reflected solar radiation, and planet-emitted radiation for the IU surface as a function of orbital position. The results were obtained in three forms: (1) a tabular listing of the heat fluxes (direct solar, earth albedo, and earth emission) as a function of true anomaly and orbit time; (2) a CRT plot of the heat fluxes versus true anomaly; and (3) punched IBM cards with heat fluxes versus time (to be used in conjunction with the NAA Thermal Analyzer program).

A comparison made of the heat fluxes for all the cases indicated that, in a number of cases, the fluxes were either identical or very close. On the basis of this review, four cases were selected as adequately representing

Table 1-1. Orbits and Vehicle Orientation Data

Case No.	Case Description	
	Orbit Description	Vehicle Orientation
IU-1	200-n. mi. circular orbit; angle of inclination = 29°; launch date approximately June 21 (right ascension - 0°)	(a) X axis tangent to flight path -Z axis earth-oriented (b) X axis sun-oriented Z axis perpendicular to orbit plane (c) Y axis sun-oriented Z axis perpendicular to orbit plane
IU-2	200-n. mi. circular orbit; angle of inclination = 34°; launch date approximately December 21 (right ascension - 0°)	(a) Same as Case IU-1-(a) (b) Same as Case IU-1-(b) (c) Same as Case IU-1-(c)
IU-3	200-n. mi. circular orbit; in plane of terminator, angle of inclination = 90°; launch date approximately March 21 (right ascension - 0°)	(a) Same as Case IU-1-(a) (b) X axis sun-oriented Z axis in orbit plane (c) Y axis sun-oriented Z axis in orbit plane
IU-4	100-n. mi. circular orbit; equatorial-angle of inclination = 0°; launch date approximately March 21 (right ascension - 0°)	(a) Same as Case IU-1-(a)
IU-5	19, 327-n. mi. (synchronous altitude) circular orbit; angle of inclination = 10° (continuous solar exposure); launch date approximately March 21 (right ascension - 0°)	(a) Y axis sun-oriented Z axis perpendicular to orbit plane (b) X axis sun-oriented Z axis perpendicular to orbit plane (c) X axis tangent to flight path -Z axis earth-oriented
IU-6	19, 327-n. mi. (synchronous altitude) circular orbit; angle of inclination = 0° (maximum shadow period); launch date approximately March 21 (right ascension - 0°)	(a) Same as Case IU-5-(a) (b) Same as Case IU-5-(b) (c) Same as Case IU-5-(c)
IU-7	100-n. mi. circular orbit; angle of inclination 29°; launch date approximately June 12 (right ascension - 0°)	(a) Same as Case IU-1-(a) (b) Same as Case IU-1-(b) (c) Same as Case IU-1-(c)
IU-8	100-n. mi. circular orbit; angle of inclination = 34°; launch date approximately December 21 (right ascension - 0°)	(a) Same as Case IU-1-(a) (b) Same as Case IU-1-(b) (c) Same as Case IU-1-(c)
IU-9	Lunar or deep space injection	(a) Same as Case IU-5-(a) (b) Same as Case IU-5-(b)
IU-10	100-n. mi. circular orbit; angle of inclination = 34°; launch date approximately December 21 (right ascension - 180°)	X axis tangent to flight path
IU-11	200-n. mi. circular orbit; angle of inclination = 34°; launch date approximately December 21 (right ascension - 180°)	X axis tangent to flight path

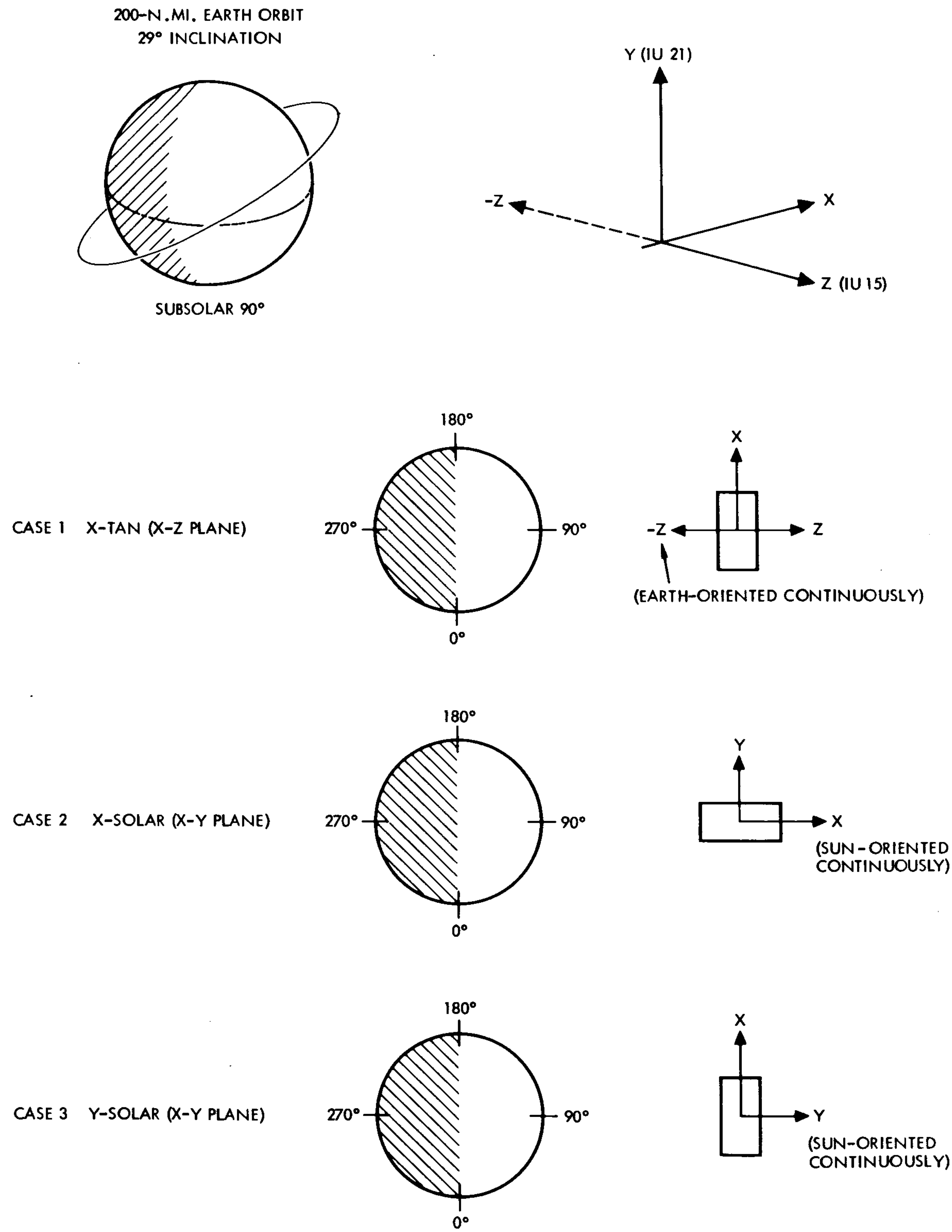


Figure 1-3. Vehicle Orbit Orientations

all the possible variations in the orbital heat fluxes on the IU. These cases contain the maximum and minimum heat fluxes. Table 1-2 indicates the four cases that were used in the thermal analysis and equivalent cases for those that were not. This essentially reduced the total number of cases to be investigated by eliminating duplications.

Table 1-2. Applicable Orbital Cases for Thermal Analysis

Case No.	Cases to Be Used	Cases Not to Be Used	Equivalent Case
IU-1	X		
IU-2		X	IU-1
IU-3	X		
IU-4	X		
IU-5		X	IU-6
IU-6	X		
IU-7		X	IU-1
IU-8			IU-2
IU-9		X	IU-6
IU-10	X		
IU-11	X		

ASTRIONIC EQUIPMENT

For the mission models selected, the functional requirements of the astrionic equipment were assumed to be unchanged. Thus, with the addition of a few new pieces of equipment, the current list of astrionic equipment was assumed to be adequate. On the basis of a modified equipment list, the heat loads and temperature tolerance ranges were established. The Saturn V IU astrionic packages perform the functions of guidance, flight control, instrumentation, and telemetry. In any extension of the life capability of the IU, therefore, it is expected that the Saturn V IU would perform at least these same functions. Astrionic equipment that presently performs these functions is described in Reference 1-1.

This section of the report does not delve into specific electronic and functional changes in the astrionic equipment needed to extend the operational life of the present Saturn V hardware, because this aspect was considered outside the scope of this essentially vehicle-related TCS study, and because studies already were in progress in this area (Reference 1-2). Reference 1-2 considers the life extension of the IU to periods of up to 30 days. A significant result of this IBM study was that very little change to the astrionic equipment was required for life extension of up to 30 days. For mission life extensions of up to 180 days, however, it is expected that drastic changes in the design of astrionic equipment are required.

As a result of microminiaturization, the following are some of the general design changes which are anticipated:

1. Heat loads of individual instrument packages will be smaller. As 50-watt packages have become more common than 200-watt packages, so will 15-watt packages become more common than 50-watt packages.

2. Total astrionic system heat loads will become only slightly smaller. The system will consist of more packages performing more functions despite consolidation.

3. The astrionic unit will depend more on forced cooling to meet higher reliability goals and to counter otherwise higher temperatures resulting from increased heat densities.

4. The integral coldplate packaging concept will become increasingly dominant. This concept is one in which the coldplate is a physical and usually a structural part of the instrument package and is contained within the package envelope.

5. Active vehicle cooling systems that supply coolant to the individual instruments will continue to be required. The larger number of smaller heat load packages will permit series and series-parallel coolant-loop connections to permit the coolant to return to the vehicle TCS at the warmest possible temperature.

The IU has been developed on a building-block concept, which allows it to be easily modified for various missions by adding or deleting various instruments and by reprogramming the launch vehicle digital computer (LVDC). This concept is expected to be continued not only for the Apollo program but also for future missions in which Saturn V hardware could be utilized. From the astrionics viewpoint, this concept allows the improvement of any instrument or equipment group independent of other instruments.

A suggested list of vehicle electronic systems to be studied (Reference 1-3) is:

1. Guidance Systems

   a. Inertial stable platforms

   b. Data adapter

   c. Digital computer

d. Horizon sensor

e. Star tracker

2. Control System

a. Rate gyros

3. Measuring and Telemetry

a. Measuring racks

4. Radio Frequency Systems

a. Radar altimeter

b. C-Band radar

c. Minitrack

d. Azusa transponder

5. Electrical Systems

a. Batteries

b. Power-distributors

c. Switch selector

6. Radar System

The items of electronic equipment listed, except for the guidance optical sensors and radars, are current IU inventory items. The thermal characteristics of this IU equipment are shown in Figure 1-4, in which the ordinate is the tolerable case temperature range of the instrument and the abscissa is the heat load of the equipment. The equipment list, temperature range, and heat load data were provided by NASA/MSFC for this study (References 1-4 and 1-5).

In Figure 1-4, the equipment is plotted in the order of a diminishing lower temperature limit. That is, batteries, which have a lower operating temperature limit of 68 F, are plotted first, and the flight control computer and the control signal processor, which have a lower operating temperature limit of -67 F, are plotted last. This readily shows the heat loads associated with the temperature sensitivity of the equipment and allows grouping of the equipment on a thermal capability basis.

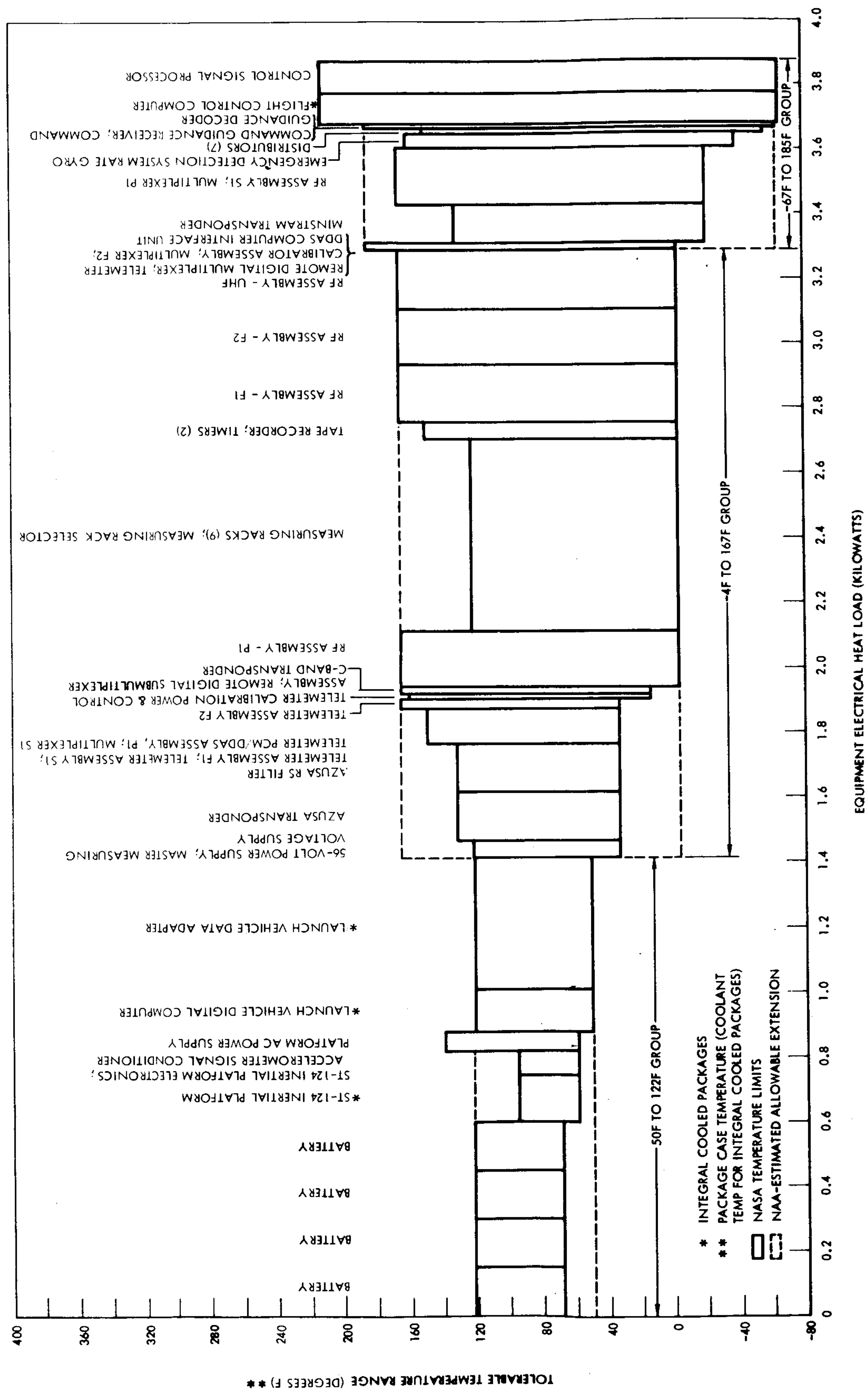


Figure 1-4. Tolerable Case Temperature Range

To minimize the number of groups, the tolerable temperature range of some packages was extended beyond that specified by NASA. In these instances, it is believed that the NASA limits are too conservative and that the wider temperature limits are inherently tolerable for the equipment (or could be made tolerable with only minor thermal design improvements within the packages). For the packages on the present Saturn V instrument unit, three groups appeared logical: 50 F to 122 F (Group I); -4 F to 167 F (Group II); and -67 F to 185 F (Group III). The packages in Group I were, for practical purposes, expected to require an active TCS utilizing a recirculating coolant. The supply temperature of the coolant would be maintained within allowable limits by a suitable temperature control system. Equipment in Group II was expected to require a similar active TCS with a recirculating coolant. However, the wider permissible temperature range of the packages in this group was considered to make active temperature control of the coolant unnecessary. This assumption was based on the belief that the vehicle coldplates, acting as thermal radiators, might transfer a sufficient quantity of heat to the instrument unit outer shell and the adjacent structure and thus maintain the coolant temperature within allowable limits. The permissible temperature range for Group III equipment is such that a completely passive thermal conditioning system appeared to be entirely feasible for packages in this group. To realize the potential of this method, advantage is taken of the thermal mass of the electronic equipment and the vehicle coldplates and/or structure on which the equipment is mounted. This thermal mass can prevent the equipment from exceeding tolerable temperature limits as environmental heat loads change with orbital positions. One exception to a completely passive TCS in Group III is the integrally cooled flight control computer. Although, from a standpoint of temperature range tolerance, this unit appears in Group III, the requirement for coolant flow would require that it be contained in Group II.

The equipment function (i. e., the generic equipment) required for the postulated missions may be grouped in the three following categories: guidance and control; measuring, RF, and telemetry; and electrical. For the purpose of this study, the primary power supply (batteries, fuel cells, solar cells, etc.) is not considered astrionic equipment and is discussed in another section. Primary mission experiments and sensors also are not included because this is special equipment whose requirements could vary widely.

Table 1-3 gives probable typical equipment characteristics for IU applications for the present operational time period (1966 to 1969), the intermediate future operational time period (1968 to 1971), and the future operational time period (1970 to 1975). The equipment type, supplier, probable package heat load, package weight, tolerable case temperature range, tolerable nonoperating temperature range, and probable cooling methods are listed in the table. The present IU equipment shown in Figure 1-4 is included under present time period, along with other probable candidate units such as the horizon sensor, star tracker, auxiliary computer memory storage, tape recorder, radar altimeter, rendezvous radar, and HF and S-band communications, which are not part of the present Saturn V IU. These probable candidate units have been included in accordance with the contract statement of work.

It should be emphasized that the equipment shown in Table 1-3 is not intended to be typical for any one mission; rather it is a complete shopping list from which a specific list can be selected for a specific mission or experiment, in keeping with the building-block concept followed by NASA/MSFC. In practice, a final list is selected for each type of mission from exhaustive tradeoff studies.

The intermediate future equipment of Table 1-3 is, for the most part, improved Saturn V hardware. The basic improvement assumed is the incorporation of integrated circuits into existing electronic circuits.

The future operational time period equipment is assumed to be predominantly newly designed, fully microminiaturized electronics. It is expected that an all-integrated circuit or similar microelectronic approach would be followed. It is believed that in this operational time period, in which most of the longer duration missions (approaching 180 days) would be made, the microelectronic approach would be mandatory on the basis of required reliability improvements. The trends and effects of microminiaturization and microelectronics are presented in Appendix 2B to Volume 2 of this report (SID 67-373-2, Pages 239-255).

Heat load, weight, temperature tolerance, and cooling method requirements for Table 1-3 equipment applicable to the present operational time period (1966-1969) are based on information supplied by NASA (References 1-4 and 1-5). Listed characteristics of equipment applicable to the two future operational time periods (1968-1971 and 1970-1975) represent best estimates based on the experience of NAA and other manufacturers of this equipment. The equipment listed in the Present column in Table 1-3 is carried over into the Intermediate Future and Future columns. This was done for comparative purposes only and does not imply that equipment functions or packages would not be consolidated. Where feasible, consolidation can be expected to minimize the number of small units. For example, a typical present IU contains nine measuring racks. The microelectronic versions of these might be packaged so that there would be only three racks, each with the capability of three present racks. As another example, the power supplies associated specifically with the ST-124 of the present system undoubtedly would become a part of the platform electronic package in the future systems.

Equipment		Present Operational Time Period (1966-1969) Equipment Type (Suppliers)	Heat Load (watts)	Weight (lb)	Case Temp Tolerance (F) (NASA Data)		Case Temp Tolerance (F) (NAA Est)		Nonoperating Temp Tolerance (F)(NAA)		Cooling Method	Equipme (Supp
Guidance & Control	Inertial navigator	SV - ST 124 plat (Bendix)	70	118	50	95	50	122	30	185	Integral	Strapdown guidar
		SV - ST 124 plat elec (Bendix)	46	40	59	95	50	122	-65	200	Veh CP	instruments, ele
		SV - Plat a-c PS (Bendix)	70	30	59	140	50	122	-65	185	Veh CP	power supply)
		SV - Bearing gas supply	5	35	-65	160	-65	160	-80	250	Veh CP	
	Computer complex	SV - LVDC (IBM)	142	88	50	122	50	122	-65	185	Integral	4 Pi type (IBM)
		SV - LVDA (IBM)	400	190	50	122	50	122	-65	185	Integral	4 Pi type (IBM)
	Flight controls	SV - Flt cont computer (ECS)	100	125	-67	212	-65	185	-80	200	Integral	Modified SV with
		SV - Rate gyros	45	10	-40	160	-65	185	-80	200	Veh CP	SV with improve
		SV - Control accel	20	12	-20	160	-40	160	-65	185	Veh CP	SV control acc
		SV - Accel sig cond (Bendix)	6	10	59	95	50	122	-65	185	Veh CP	Improved SV
		SV - Control sig proc (Martin)	100	38	-67	212	-65	185	-80	200	Veh CP	Micromin SV
	Flight sequencer	SV - Timers	7	1	- 4	140	- 4	165	-65	185	Veh CP	Apollo flt sequer
	Sensors	Horizon sensor - OGO	10	17	—	—	0	140	-40	160	Veh CP	Horizon sensor (
		Star tracker - lunar orb (ITT)	15	10	—	—	-65	160	-65	185	Veh CP	Star tracker - lu
	Aux memory store	Miniature computer memory (IBM)	20	2.5	0	140	50	122	-40	160	Veh CP	NDRO cove mem
	Flight sequencer	SV - Selector switch (ECI)	0	20	-13	175	-65	180	-80	200	Veh CP	Improved SV
Measuring, RF and Telemetry	Telemetry	SV - TM - F1	35	17.5	32	149	- 4	167	-65	200	Veh CP	Improved SV
		SV - PCM/DDAS, - P1	60	27	32	149	- 4	167	-65	200	Veh CP	Improved SV
		SV - TM - S1	8	17.5	32	149	- 4	167	-65	200	Veh CP	Improved SV
		SV - TM - F2	30	14	32	149	- 4	167	-65	200	Veh CP	Improved SV
	Telemetry RF	SV - RF assy - F1	177	13.4	- 4	167	- 4	167	-65	200	Veh CP	Improved SV
		SV - RF assy - PCM	177	15	- 4	95	- 4	167	-65	200	Veh CP	Improved SV
		SV - RF assy - S1	177	13.4	-22	167	-65	185	-65	200	Veh CP	Improved SV
		SV - RF assy - F2	177	13.4	- 4	167	- 4	167	-65	200	Veh CP	Improved SV
		SV - RF assy - VHF	180	11	- 4	167	- 4	167	-65	200	Veh CP	Improved SV
	Telemetry accessories	SV - TM calib	4.9	5.3	- 4	185	-65	185	-65	200	Veh CP	Improved SV
		SV - TM calib control	5.6	3.5	14	185	- 4	167	-65	200	Veh CP	Improved SV
	Signal conditioner	SV - Meas rack	60	21	- 4	122	- 4	167	-65	200	Veh CP	New micromin c
		SV - Meas rack sel	5	2.4	- 4	122	- 4	167	-65	200	Veh CP	SV with improve
	Multiplexer	SV - P1 multiplexer	6	21.3	-22	185	-65	185	-80	200	Veh CP	New micromin c
		SV - Remote dig.	3	13	- 4	185	-65	185	-80	200	Veh CP	New micromin c
		SV - Remote dig. sub	4	13.2	14	158	- 4	167	-80	200	Veh CP	New micromin c
		SV - Slow speed	14	12.5	32	149	- 4	167	-80	200	Veh CP	New micromin c
		SV - F2	6	21.3	- 4	185	-65	185	-80	200	Veh CP	New micromin c
	Miscellaneous equipment	SV - Tape recorder	40	10	- 4	149	- 4	167	-40	185	Veh CP	Apollo tape reco
		SV - DDAS interface unit	3	11.6	- 4	185	-65	185	-80	200	Veh CP	SV - DDAS inter
		SV - Coaxial switch	7	1	—	—	-65	185	-80	200	Veh CP	SV - coaxial sw
		Lunar mod TV camera (Westhse)	7	8	—	—	- 4	167	-40	185	Veh CP	Lunar mod TV c
	Radars	SV - C-Band Xponder (Motorola)	24	5.5	14	167	- 4	167	-65	200	Veh CP	SV C-band tran
		SV - Radar altimeter (Westhse)	65	15	—	—	- 4	167	-65	200	Veh CP	Improved SV ra
		Gemini rendezvous radar (Westhse)	65	15	—	—	- 4	167	-65	200	Veh CP	Improved Gemi
	Radio command	SV Command receiver	4	3	-65	185	-65	185	-65	200	Veh CP	SV command re
		SV Command decoder	4.5	6	-65	185	-65	185	-65	200	Veh CP	SV command de
	Range tracking	SV Azusa transponder	135	19	32	131	- 4	167	-65	200	Veh CP	Improved SV
		SV Minstram transponder	140	16.5	-22	131	- 4	167	-65	200	Veh CP	Improved SV
		SV Azusa RI filter	135	10	32	131	- 4	167	-65	200	Veh CP	Improved SV
	Communications	Apollo Uni-S band (Collins)	18	38	—	—	0	150	-65	185	Veh CP	Apollo Uni-S-ba
		Apollo - HF transceiver	50	7	—	—	0	150	-65	185	Veh CP	Improved HF tr
Electrical	Primary power source	SV Battery (Eagle Picher)	150	165	68	122	50	122	-65	160	Veh CP	
		Apollo fuel cell (P&W)										
		Solar cells										
		Radioisotopic thermal gen										
	Secondary power supply	SV 56-v power supply	50	9.5	32	122	- 4	167	-65	200	Veh CP	
		SV master meas voltage	7	2	32	122	- 4	167	-65	200	Veh CP	Improved SV
	Distributors	SV (various types)	5	30	-58	149	-65	185	-65	200	Veh CP	Improved SV

Table 1-3. Astrionic Equipment Lists

Intermediate Future Operational Time Period (1968-1971)						Future Operational Time Period (1970-1975)					
nt Type ier)	Heat Load (watts)	Weight (lb)	Case Temp Tolerance (F) (NAA Est)	Nonoperating Temp Tolerance (F)	Probable Cooling Method	Equipment Type (Supplier)	Heat Load (watts)	Weight (lb)	Case Temp Tolerance (F) (NAA Est)	Nonoperating Temp Tolerance (F)	Probable Cooling Method
ce(inertial	150	100	0 125	-65 200	Veh CP	Improved strapdown guidance	50	50	0 125	-65 200	Integral
tronics, and	—	—	— —		—	(inertial instrument:, elec-	—	—	— —	— —	—
	—	—	— —		—	tronics, and power :upplies)	—	—	— —	— —	—
	—	—	— —		—		—	—	— —	— —	—
	70	20	30 125	-65 185	Integral	Advanced micromin	25	10	30 125	-65 185	Integral
	200	40	30 125	-65 185	Integral	Advanced micromin	60	20	30 125	-65 185	Integral
IC's	50	75	-65 185	-80 200	Integral	Advanced micromin.dig	25	30	-65 140	-80 200	Integral
nents	40	10	-65 185	-80 200	Veh CP	SV with improvements	40	10	-65 185	-80 200	Veh CP
	20	12	-40 160	-65 185	Veh CP	SV control acc	20	12	-40 160	-80 200	Veh CP
	4	7	0 125	-65 185	Veh CP	Improved SV	4	7	0 125	-80 200	Veh CP
	30	15	-65 140	-80 200	Integral	Micromin SV	30	15	-65 140	-80 200	Integral
er (NAA)	7					Apollo flt sequencer (NAA)	7				
OGO)	10	17	0 140	-40 160	Veh CP	Horizon sensor - impr OGO	8	12	-65 160	-40 160	Veh CP
nar orb (ITT)	15	10	-65 160	-65 185	Veh CP	Star tracker - lunar orb (ITT)	15	10	-65 160	-65 185	Veh CP
ry	5	5	0 140	-40 160	Veh CP	NDRO core memory	5	5	0 140	-40 160	Veh CP
	0	10	-65 185	-80 200	Veh CP	Improved SV	0	10	-65 175	-80 200	Veh CP
	20	10	-4 167	-65 200	Veh CP	New micromin	5	5	-65 140	-80 200	Integral
	40	17	-4 167	-65 200	Veh CP	New micromin	10	10	-65 140	-80 200	Integral
	8	10	-4 167	-65 200	Veh CP	New micromin	2	5	-65 140	-80 200	Integral
	20	10	-4 167	-65 200	Veh CP	New micromin	5	5	-65 140	-80 200	Integral
	50	10	-4 167	-65 200	Veh CP	Improved SV	50	10	-4 167	-65 200	Veh CP
	50	10	-4 167	-65 200	Veh CP	Improved SV	50	10	-4 167	-65 200	Veh CP
	50	10	-4 167	-65 200	Veh CP	Improved SV	50	10	-4 167	-65 200	Veh CP
	50	10	-4 167	-65 200	Veh CP	Improved SV	50	10	-4 167	-65 200	Veh CP
	50	10	-4 167	-65 200	Veh CP	Improved SV	50	10	-4 167	-65 200	Veh CP
	4	4	-65 185	-65 200	Veh CP	Improved SV	4	4	-65 185	-65 200	Veh CP
	4	2	-65 185	-65 200	Veh CP	Improved SV	4	2	-65 185	-65 200	Veh CP
ign	20	8	-4 167	-65 200	Veh CP	Advanced micromin	10	5	-65 140	-80 200	Integral
ents	4	2	-4 167	-65 200	Veh CP	SV with improvements	4	2	-4 167	-65 200	Veh CP
ign	4	4	-65 160	-80 200	Integral	New micromin	4	4	-65 160	-80 200	Integral
ign	4	4	-65 160	-80 200	Integral	New micromin	4	4	-65 160	-80 200	Integral
ign	4	4	-65 160	-80 200	Integral	New micromin	4	4	-65 160	-80 200	Integral
ign	4	4	-65 160	-80 200	Integral	New micromin	4	4	-65 160	-80 200	Integral
ign	4	4	-65 160	-80 200	Integral	New micromin	4	4	-65 160	-80 200	Integral
er (Leach)	26	22	-4 167	-65 185	Veh CP	Improved Apollo tape recorder	20	15	-4 167	-65 185	Veh CP
e unit	3	11.6	-65 185	-80 200	Veh CP	SV DDAS interface unit	3	11.6	-65 185	-80 200	Veh CP
h	7	1	-65 185	-80 200	Veh CP	SV coaxial switch	7	1	-65 185	-80 200	Veh CP
ers (Westhse)	7	8	-4 167	-40 185	Veh CP	Lunar mod TV camera	7	8	-4 167	-40 185	Veh CP
nder	24	5.5	-4 167	-65 200	Veh CP	Improved SV C-band Xponder	15	5	-4 167	-65 200	Integral
lt	40	25	-4 167	-65 200	Veh CP	Improved SV rad alt	40	25	-4 167	-65 200	Veh CP
endez rad	40	10	-4 167	-65 200	Veh CP	Improved Gemini rendez rad	40	10	-4 167	-65 200	Veh CP
ver	4	3	-65 185	-65 200	Veh CP	SV command receiver	4	3	-65 185	-65 200	Veh CP
er	4.5	6	-65 185	-65 200	Veh CP	SV command decoder	4.5	6	-65 185	-65 200	Veh CP
	80	15	-4 167	-65 200	Veh CP	Improved SV	80	15	-4 167	-65 200	Veh CP
	80	12	-4 167	-65 200	Veh CP	Improved SV	80	12	-4 167	-65 200	Veh CP
	80	8	-4 167	-65 200	Veh CP	Improved SV	80	8	-4 167	-65 200	Veh CP
	18	38	0 150	-65 185	Veh CP	Improved Uni-S-band	15	20	-4 167	-65 200	Veh CP
ceiver	40	7	-4 167	-65 185	Veh CP	Improved HF transceiver	40	7	-4 167	-65 185	Veh CP
	5	2	-4 167	-65 200	Veh CP	Improved SV	5	2	-4 167	-65 200	Veh CP
	5	20	-65 185	-65 200	Veh CP	Improved SV	5	20	-65 185	-65 200	Veh CP

The selection of a strapdown inertial navigator for the intermediate future and future merits further discussion. The development of strapdown inertial navigators is such that, with the use of a horizon sensor, attitude reference in the order of ±1 degree is readily achieved. More precise reference or specific mission/experiment requirements may necessitate a gimbaled platform, perhaps with stellar supervision (star tracker). If the ST-124 is considered for this gimbaled platform, the gas-bearing nitrogen supply requirement imposes severe penalties. A closed-loop (recirculating) nitrogen supply, although mandatory for mission durations of approximately 24 hours, is not a satisfactory solution. A more reasonable solution would be to replace the three single-axis gyros of the ST-124 with two self-lubricating, spherical gas-bearing gyros.

Table 1-4 gives the probable equipment quantity per system and is intended to show the increased equipment quantity (complexity) with increased mission duration. The indicated increases are due primarily to the need for backup and redundancy. Functional requirements remain fairly constant with mission duration (i.e., longer missions do not necessarily imply increased functions). The quantities shown in Table 1-4 for durations other than up to 12 hours are relative to the quantities indicated in the 12-hour column. For example, two LVDC's are indicated for mission durations greater than 30 days while only one is required for a 12-hour mission. This does not mean that there will physically be two computers in the longer missions (although there may be), but it does mean that the computer for the longer duration is about twice the complexity and capacity of the first, due primarily to redundancy requirements. These quantities will affect the total vehicle astrionic heat load, discussed in another section.

The discussion on microelectronics (Appendix 2B in Volume 2) indicates that heat loads for the same functional requirements will continually decrease. At the present pace of microelectronics, a conservative estimate is that the total electrical heat load will decrease perhaps by half every two years; but with continually increasing functional requirements and astrionic system complexity (made possible and attractive by smaller size, weight, and cost, and improved reliability), the total heat load may decrease more slowly, perhaps halving every four or five years. This seemingly slow improvement is dictated by economic and other factors rather than by technological capability.

Much equipment, particularly of the digital circuitry type, will decrease at the much faster rate. Other equipment, such as the transmitter (output) stage of radars and RF units, which depend on transmitted power, will not decrease as significantly. Even here, however, available techniques—such as trading off digital storage of information against transmitting information in high-frequency, short-duration bursts—will be used to reduce heat dissipation.

Table 1-4. Probable Equipment Quantity Per System Requirement*

Equipment	Mission Duration					
	To 12 hr	To 48 hr	To 10 days	To 30 days	To 90 days	To 180 days
Inertial navigator	1	1	1	2**	2**	2**
Launch vehicle digital computer	1	1	1	2	2**	2**
Launch vehicle data adapter	1	1	1	1	1	1
Auxiliary memory storage	0	0	1	1	1	1
Flight control computer	1	1	1	2	2	2
Rate gyro	3	3	3	3	3	3
Control accelerometer	2	2	2	2	2	2
Accelerometer signal conditioner	1	1	1	1	1	1
Control signal processor	1	1	1	1	2	2
Flight sequencer	2	2	2	2	2	2
Horizon sensor	1	1	1	1	1	1
Star tracker	1	1	1	1	1	1
Selector switch	1	1	1	2	2	2
Telemetry	4	5	6	7	8	10
Telemetry RF	5	6	7	8	9	11
Telemetry signal conditioner	9	9	9	10	12	14**
Telemetry multiplexer	6	7	8	9	10	12
Tape recorder	0	0	1	1	1	2
TV camera	0	0	0	1	1	1
C-band transponder	1	1	0	0	0	0
Radar altimeter	1	1	1	1	1	1
Rendezvous radar	0	1	1	1	1	1
Radio command	1	1	2	2	2	2
Range tracking	2	2	2	2	2	2
S-band communication	0	0	1	1	1	2
Distributor	5	5	5	5	6	8
Special power supplies	7	1	7	2	2	2

*Includes backup and redundancy requirements
**Present operational time period equipment not practical for long-duration mission

Table 1-5 is a comparison of a modern digital computer, the D-37C computer in production for Minuteman II, with more advanced computers that can be available within the next two to six years. Heat dissipation is reduced from about 300 to 15 watts. It was recently found that a digital computer to perform the hypothetical space vehicle computation requirements shown in Table 1-6 could be built by conventional microelectronic techniques to dissipate only 12 to 15 watts. The hypothetical mission was a composite of missions studied for standardized space guidance system (SSGS) for the USAF having the mission phases of (1) prelaunch, (2) atmospheric ascent, (3) exo-atmospheric ascent, (4) orbital coast, (5) orbital change, (6) rendezvous, (7) de-orbit, and (8) reentry. By using a distributed logic computer system (Reference 1-6) it was found that the same computational requirements could be implemented with dissipation of only 5 to 6 watts.

Table 1-7 indicates the equipment required to be completely or partly operative during various phases of a typical mission. These phases include (1) preflight, (2) ascent, (3) orbital checkout, (4) orbit transfer, (5) parking orbit full power on, (6) parking orbit standby, (7) deactivated, (8) parking orbit rendezvous, and (9) translunar or planetary injection. Orbital checkout is concerned with the checkout of on-board equipment while in orbit prior to subsequent use. Parking orbit standby includes the operation of equipment necessary to reactivate other on-board equipment and to maintain attitude control. In parking orbit deactivated, only equipment necessary to reactivate other equipment remains operative, and attitude is not controlled. The other mission phases are self-explanatory.

Table 1-5. Computer Characteristics Summary

	D-37C	Advanced Computer		
Availability	Production Item	1 year	2 to 3 years	3 to 5 years
Weight (lb)	39	19	13	9
Volume (cu ft)	0.73	0.26	0.13	0.08
Dimension (in.)	20.9x5.7x10.5	5-1/2x7-1/2x11	5-1/2x7-1/2x5-1/2	4-1/2x6-1/2x4-1/2
Power (watts)	300	72	45	15
Clock Rate (kc)	345.6	750	750 2 ϕ	750 4 ϕ
Add Time (sec)	78.25	4	4	4
Memory Capacity:				
words	7222	17,408	17,408	17,408
bits	24	24	24	24
Technology	Integrated circuits, disc memory	MOS IC's, laminated ferrite memory	Polytab MOS IC's, laminated ferrite memory	Reg bond MOS IC's, epitaxial ferrite memory

Table 1-6. General Purpose Computer Requirements

	Storage, Instructions, and Constants	Operations per Mission	Seconds per Iteration	Operations per Second
Prelaunch				
Accelerometer calibration	520	660	Not Critical	
Platform alignment and gyro bias calculation	340	540	Not Critical	
Mission targeting	380	340	300	1
Subtotal	1240			1
Atmospheric Ascent				
IMU mechanization	320	280	0.1	2800
Navigation computation	290	440	0.1	4400
Steering	220	170	0.1	1700
Subtotal	830			8900
Exo-atmospheric Ascent				
IMU mechanization	320	280	0.1	2800
Navigation computation	290	440	0.1	4400
Attitude control	290	70	0.1	700
Subtotal	900			7900
Orbit Coast				
Integrate free-fall navigation equations of motion	60	1800	10	180
Orbit determination by smoothing and filtering	4660	8700	10	870
Sensor pointing	360	1930	0.33	5790
IMU alignment and gyro bias calculation by smoothing and filtering	80	430	Not Critical	
Subtotal	5160			6840
Orbit Change				
Set up rendezvous injection	1280	1940	300	7
Set up midcourse maneuver	220	8000	60	14
IMU mechanization	320	280	0.2	1400
Navigation computations	290	440	0.2	2200
Orbit change steering	470	210	0.1	2100
Subtotal	2580			5721
Rendezvous				
Rendezvous sensor pointing	200	1930	0.1	1930
IMU mechanization	320	280	0.5	560
Navigation computation	290	440	0.5	880
Rendezvous vehicle control	370	490	0.1	4900
Subtotal	1180			8270
Deorbit				
Set up deorbit	690	14,870	30	496
IMU mechanization	320	280	0.5	560
Navigation computations	290	440	0.5	880
Optimal estimating	2650	14,930	30	498
Subtotal	3950			2434
Reentry				
IMU mechanization	320	280	1	280
Navigation computation	290	440	1	440
Reentry vehicle control	620	1130	0.2	5650
Subtotal	1230			6370

Table 1-7. Equipment Required to be Operative

Equipment	Preflight	Ascent	Orbital Checkout	Orbit Transfer	Parking Orbit Full on	Parking Orbit Standby	Parking Orbit De-activated	Parking Orbit Rendezvous	Translunar or Planetary Injection
Inertial navigator	x	x	x	x	x			x	x
LV digital computer	x	x	x	x	x			x	x
LV data adapter	x	x	x	x	x			x	x
FC computer	x	x	x	x	x	x		x	x
Rate gyros	x	x	x	x	x	x		x	x
Control accelero	x	x							
Accelero sig	x	x							
Cont sig proc	x	x							
Flight sequencer	x	x	x	x	x			x	x
Horizon sensor	x		x		x	x			
Star tracker	x		x	x	x				x
Aux mem store	x		x		x				
Selector switch	x	x	x	x	x			x	x
TM	x	x	x	x	x	x	x	x	x
TM RF	x	x	x	x	x	x	x	x	x
TM sig cond	x	x	x	x	x	x	x	x	x
TM multiplexer	x	x	x	x	x	x	x	x	x
Tape recorder	x	x	x	x	x			x	x
TV camera	x	x	x		x				
C-band Xponder	x		x		x				
Radar altimeter	x		x		x				
Rendezvous radar	x							x	
Radio command	x	x	x	x	x	x	x	x	x
Range tracking	x	x	x	x	x			x	x
S-band command	x	x	x	x	x			x	x
Distributors	x	x	x	x	x			x	x
Spec power sup	x	x	x	x	x			x	x

On the basis of Tables 1-3, 1-4, and 1-7, the maximum heat loads for various mission phases and several mission durations were determined. Results are shown in Figures 1-5, 1-6, and 1-7 for the present, intermediate future, and future operational time period equipment, respectively. Mission durations of 10 hours, 48 hours, 10 days, 30 days, 90 days, and 180 days are represented. It should be noted that the heat loads are maximum or, more properly, worst case. They assume that the full shopping list of equipment is utilized, i.e., they do not account for the possibility that specific equipment may be deleted for specific missions. Further, they assume that all redundant and backup equipment is energized, except for that of the orbit standby and orbit deactivated phases. The figures clearly illustrate the downward trend in equipment heat load that might be expected using future equipment and the increase in total integrated heat load with increased mission time. This latter increase stems from the requirement, for reliability purposes, for increasing quantities of redundant equipment with increase in mission duration, and from the assumption that all such equipment is energized. The figures also facilitate establishment of a heat load profile to correspond to a mission profile with various mission phases and times. This is illustrated in Figure 1-8, which represents a maximum heat load case. The drop in equipment heat load at a mission time of about 10 hours corresponds to the condition shown in Figure 1-5 for the orbital transfer period, during which some of the equipment was assumed to be turned off. Similar heat load profiles established for the three mission models selected for this study give maximum and minimum conditions, as well as heat load variations with mission phases.

The thermal design of astrionic equipment packages is affected by the means employed for transferring package heat dissipation to a heat sink. Heat transfer techniques used within astrionic packages include:

- Conductive materials
- Encapsulants
- Component fins
- Module retainers
- Module heat shunts
- Convection techniques
- Ebullient cooling
- Heat pumps
- Radiation

The application of these techniques is described in Appendix 2B to Volume 2 of this report (SID 67-373-2, Pages 255-262); a discussion of suitable temperature control devices/techniques for individual packages or coldplates may be found in the same section of the report (Pages 263-269). A summary of cooling devices and systems is shown in Table 1-8, and a more detailed description of applicable heat sink concepts is presented in Appendix 3 to

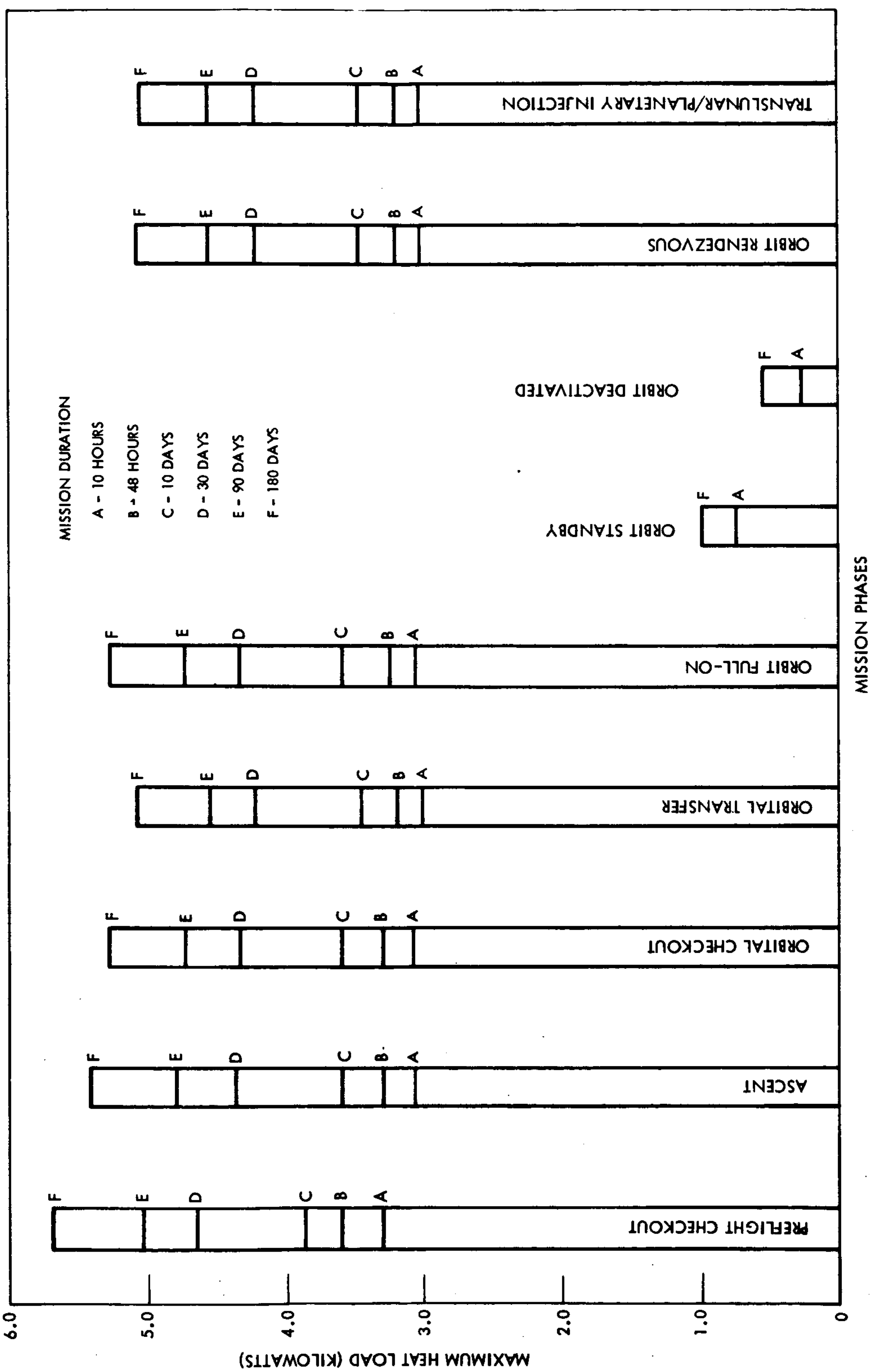


Figure 1-5. Astrionic Equipment Heat Load for Mission Phases (Operational Time Period 1966-1969)

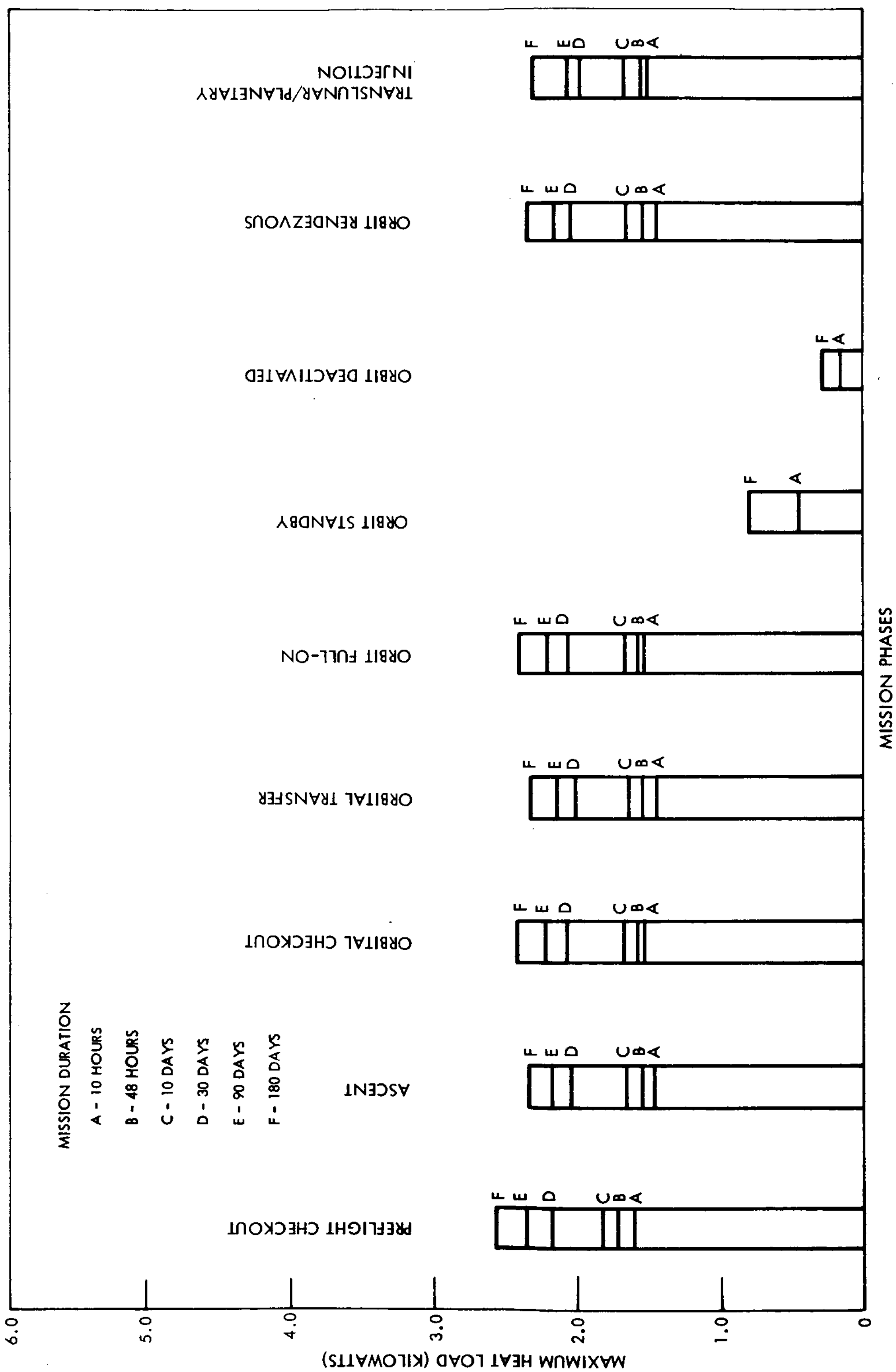


Figure 1-6. Astrionic Equipment Heat Load for Mission Phases (Operational Time Period 1968-1971)

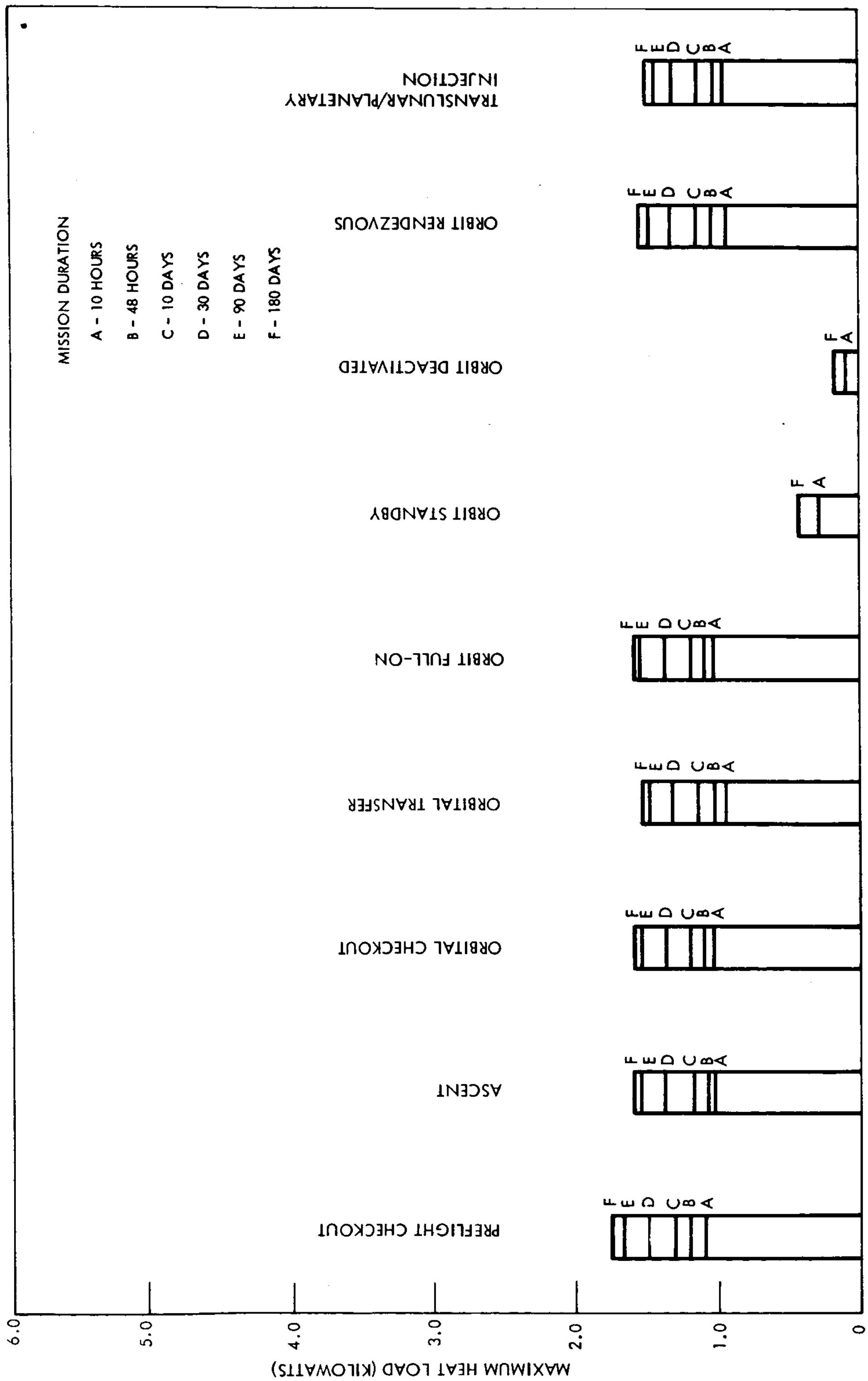


Figure 1-7. Astrionic Equipment Heat Load for Mission Phases (Operational Time Period 1970-1975)

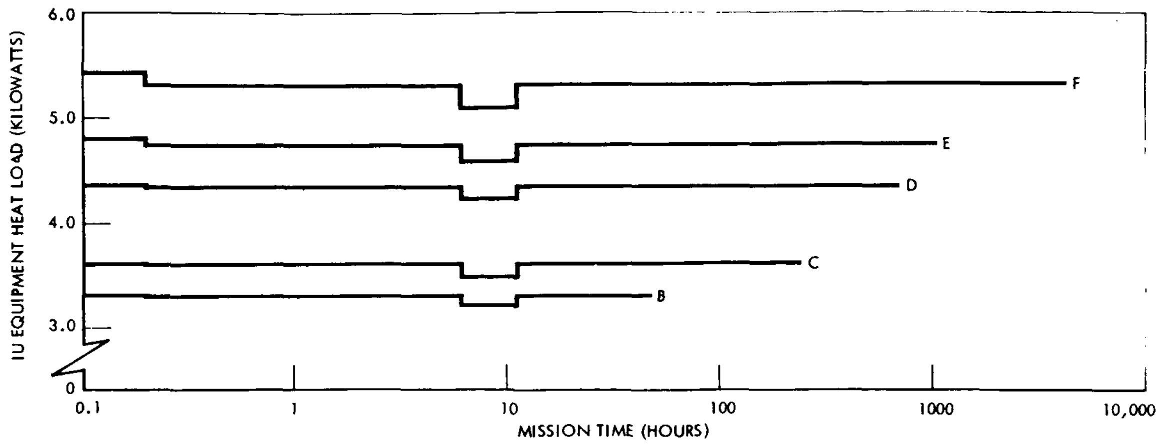


Figure 1-8. Synchronous Orbit Maximum Heat Load (Present Equipment)

Volume 2 of this report (SID 67-373-2, Pages 293-297). The regions of applicability for the cooling devices and systems listed in Table 1-8 are based on the information contained in Figure 3A-1 in Volume 2. Unidentified areas in this figure represent regions in which no single heat sink has a clear advantage over any other method. Hence, the optimum technique to be employed for mission durations and electrical heat loads in these regions will have to be established through tradeoff studies.

SPACE POWER SYSTEMS

Space power systems have been evaluated for applicability to IU missions based on 1968 hardware. Power system alternatives included radioisotopic thermoelectric generators (RTG), fuel cells, solar cells and rechargeable batteries, combined operation of fuel cells and solar cells, concepts of solar cell array integration with the Saturn vehicle, and primary batteries (for missions of short duration, such as less than one day). Space power systems of less than a one-kilowatt to four-kilowatt rating have been evaluated for applicability to IU missions on the basis of (1) power-service potentiality, (2) specific weight and service time, and (3) parametric comparisons among those power systems applicable to the missions.

For short-duration missions, primary batteries provide the best service at the least expense; for the longer missions, fuel cells, RTG, and solar cells and batteries are useful alternative power systems, as indicated in Figure 1-9. No attempt has been made to select the specific power system for the various mission periods. Rather, the intent was to indicate the leading contenders on the basis of service time and power levels. For this study, the prime interest is the thermal interface between the power system and the environmental control system.

Table 1-8. Summary of Cooling Devices and Systems

Description	Region of Applicability			Limiting Factors	Advantages and Disadvantages
	Elec. Heat Load (kw)		Mission Time (Hours)		
	Max	Min			
Thermal mass	0.06	0.01	0 - 0.05	Suitable only for extremely small power dissipation and/or short-duration peak-power loads	Can be regenerated (temperature reduced to original value when load drops)
Heat of fusion	10.0	0.01	0 - 0.60	Few suitable substances available with fusion temperature in range of interest; limited to short-duration power spikes	Can be regenerated (substance is resolidified when heat source is turned off)
Sublimators and evaporators (expendables)	10.0	0.10	0.20 - 20	Weight of expendables becomes excessive for all but short-duration missions	Utilizes high latent heat of H_2O; possible control problem during start-up for spike-type loads and under conditions of load variation
Cryogenics (residual H_2)	10.0	0.02	0.10 - 50	Availability is uncertain (depends on intended utilization of spent stage); duration depends on quantity of residuals and heat transfer into tank	Requires design of heat exchanger and control system to prevent freezing of coolant; extremely high heat capacity per unit mass
Passive radiators	0.05	0.01	More than 100	Limited to constant heat loads or equipment with large temperature tolerance	No moving parts; no power requirement; no meteoroid puncture problem; weight is not affected by mission duration
Semiactive radiators	0.60	0.10	More than 75	Required operating temperature level for efficient operation may be too high	No moving parts; no power requirement; provides some self-regulation; may pose meteoroid penetration problem; weight is not affected by mission duration
Active radiator systems	10.0	0.30	More than 15	Radiator designed for high heat load may produce freezing of fluid in tubes when load drops to standby condition	Provides temperature regulation under conditions of heat load variation; requires constant supply of power; reliability affected by meteoroid puncture probability and pump service life
Thermoelectric devices	0.15	0.01	More than 15	Suitable for thermal control of individual packages rather than complete system	Can provide heating as well as cooling; no moving parts — excellent reliability; high degree of temperature control available; high input power requirement; weight is not affected by mission duration

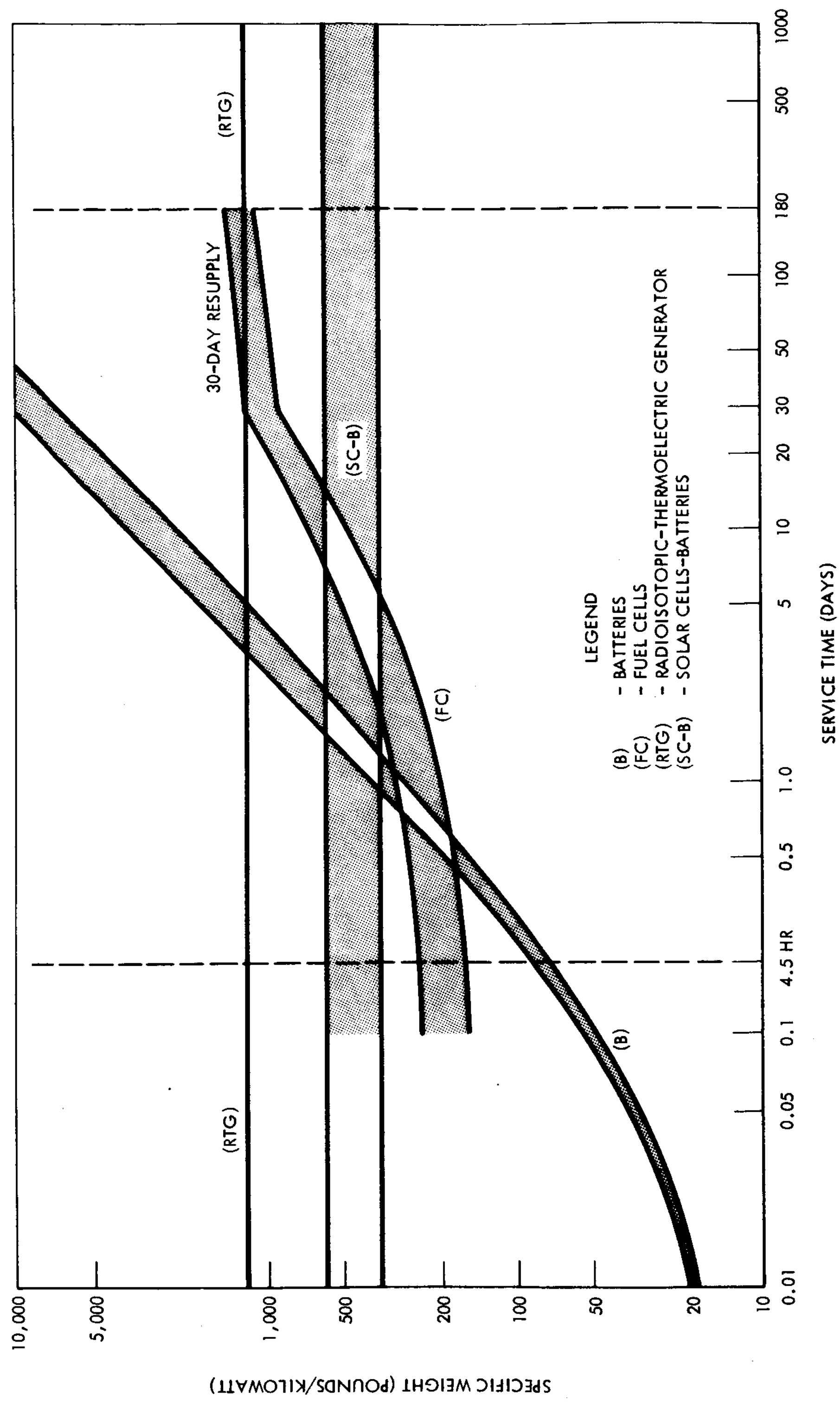


Figure 1-9. Specific Weight-Service Time Characteristics of Space Power Systems (Applicable up to 4.0 Kilowatt Output Capacity)

The thermal interfaces between the candidate power systems and the ECS are summarized in Table 1-9. The interfaces with all candidate systems are indicated since batteries are considered an integral part of all systems, either as a primary power source for short duration or as peaking batteries.

The battery cooling requirements and temperature range have been estimated for the various applications. They are similar to the current thermal requirements of the IU batteries. Thus, the present coldplate method for thermal-conditioning the batteries is considered to be adequate. The other primary power sources are generally designed with their own thermal conditioning system. The possible interfacing with a separate thermal conditioning system varies with each of the primary sources, which have been briefly reviewed.

For the fuel cell system, which is internally cooled, the indicated coolant temperature ranges and heat loads place it within Group II of Figure 1-4; thus, the system can be readily integrated into the ECS of the IU. The fuel cell or cells can be placed directly into the thermal conditioning loop or placed in a separate loop and thermally connected to the other thermal loop with a common heat exchanger. An additional interface would be the water from the fuel cell system, which could be used for cooling.

For the radioisotopic thermoelectric generator system, the self-contained or integrally designed thermal conditioning system is the only practical approach. The only point of interfacing may be the utilization of the waste heat through a heat exchanger in the RTG radiator heat rejection loop.

Solar cells, in general, would not have direct interface with an ECS in the IU except for the possible thermal interface between the radiating surfaces (passive radiators) of the solar cell panels and the IU outer surface, which would also be a radiating surface. The solar cell operating temperatures and heat loads are, in general, nominal, and so the influence of this thermal interface can be assumed to be minimal.

Table 1-9. Comparison Matrix of Two- to-Four-Kilowatt Alternative Power Systems for Instrument Unit

Parameter	Radioisotopic Thermoelectric Generator	Solar Cells Batteries	Fuel Cells
State of the art	Technology	Hardware	Hardware
Reliability	0.90	0.99	0.99 (at 50 percent redundancy)
Specific weight, lb/kwe	1160 ±100	500±125	1060 ±120
Specific shield weight, lb/kwe (5-10 rem/180 days at 16.0 ft)	205 ±15	DNA	DNA
Specific volume, ft3/kwe	11.5 ±1.0	DNA	33 ● 5
Time rate of specific volume, ft3/kwe x 30 days	DNA	DNA	33 ● 5
Specific area, ft2/kwe	125 ±20	100	20 ±5
Environmental effects	Insignificant	Radiation sensitive	Insignificant
Orientation constraints	None	±25 deg to sun 10 percent loss	None
Major advantages	Continuous radiation from heat source; integrates readily with vehicle	No refueling	Water produced, lb/day (2.0 kw) 43 (4.0 kw) 87
Major disadvantages	Loss of heat with time; crew receives radiation	Orientation requirements; atmospheric drag at altitudes below 200 n.mi.	Crew must operate fuel cell; cyrogenic tanks require refueling, typically at 30-day intervals
Orbit altitude attitude inclination effects	None	Sun eclipse; thermal effects on power output	None
Efficiency, percent	4.6 ±.5	9.0-10.0	50-65
Source specific thermal power, kwt/kwe	19.6-24.4	DNA	DNA
Specific thermal power radiated, kwt/kwe	188 ±1.9	DNA	0.65-0.70

DNA: Does not apply
*Assumes 10 percent heat loss to vehicle structure

2.0 SYSTEM INTEGRATION

The discussion in the previous section has indicated that continuing improvements in the design of electronic equipment can be expected, but a radical change in the IU as a result of these improvements may not be necessary. For the near-term application, some electronic packages may be redesigned to take advantage of current technology, which could increase the temperature tolerance, as indicated in Figure 1-3. In addition, the reliability requirement for the longer-duration missions may necessitate the use of redundant components or parts; thus, improved components or parts would be incorporated without increasing either the power requirement (or heat load) or the size of the electronic packages. A downward trend in equipment heat loads, smaller package dimensions, and increased use of integrally or internally cooled thermal design are indicated for the future, as is a relaxation in the temperature tolerances. These trends lessen the requirements on the thermal conditioning system (TCS), and so the system should tend toward a simpler design. One example is the possible change in the design of the inertial platform that would eliminate the need for a thermally conditioned nitrogen gas supply for the gas bearings. This is particularly significant for the long-duration missions.

Possible changes in the power supply can be assumed primarily for the longer missions. As mission time increases, the batteries would be replaced by other, more efficient power to serve as the primary supply; in all cases, however, batteries appear necessary to accommodate the peak loads. The change from batteries to fuel cells, for example, requires not only that a change be made in the thermal conditioning requirements, but also that the possibility of utilizing the byproduct (water) of the fuel cell be considered in the synthesis of the TCS.

In addition to the increase in mission time, a change in the functional utilization of the IU may be possible, such as the addition of experimental equipment or additional astrionic equipment. In the absence of more specific data, it has been assumed that the environmental conditioning requirements, primarily heat loads and temperature tolerances, would not be significantly different from the values indicated in the previous section. Furthermore, new items of equipment that may be added and that have a specific requirement(s) would be designed with their own TCS or within the limitations of the overall thermal conditioning system.

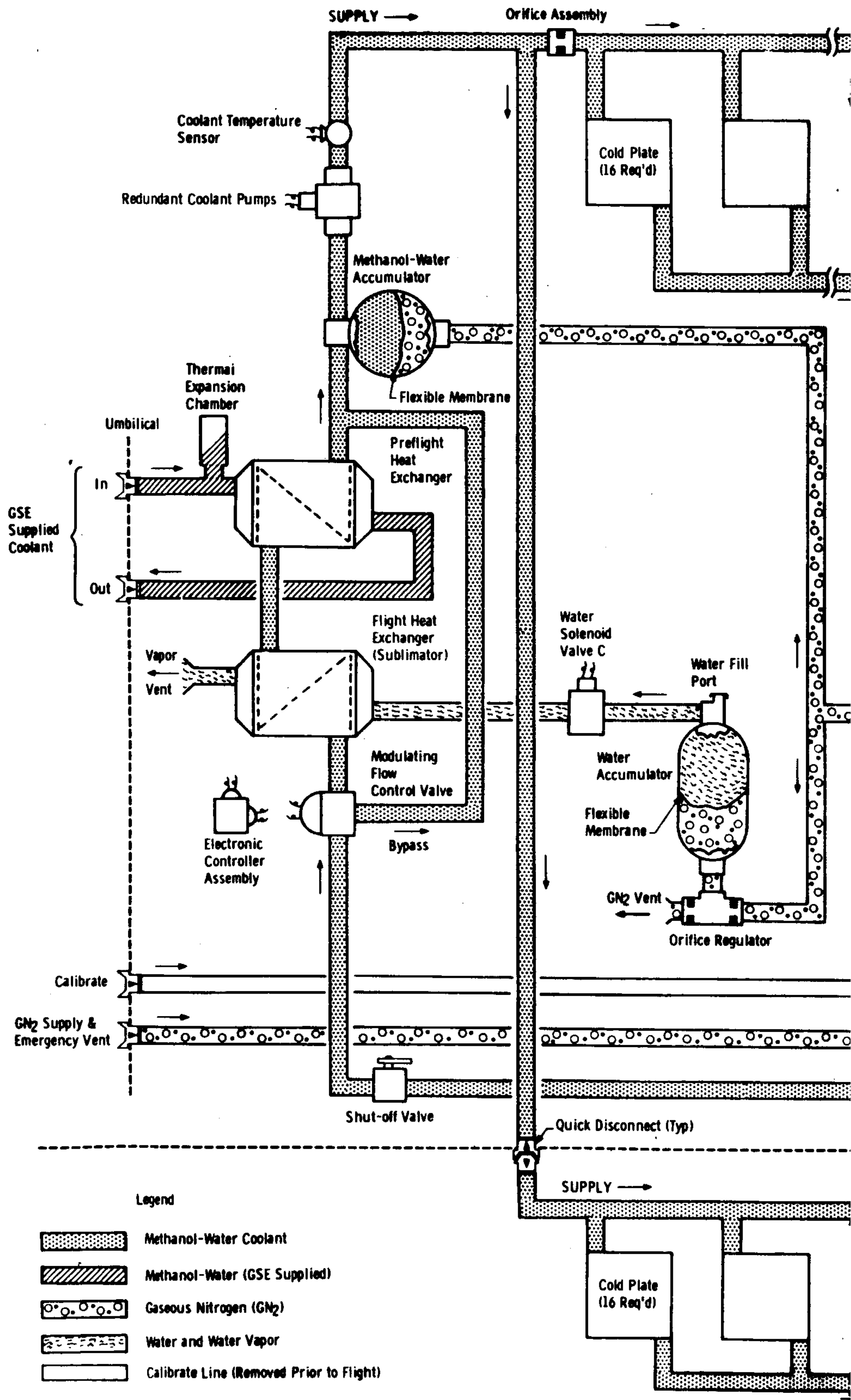
On the basis of the present and indicated future trend of the astrionic equipment and its requirements, the current TCS for the IU has sufficient merit to be considered as an initial concept, even though it has limited operational life. From a conceptual standpoint, the present arrangement of the thermal conditioning panels around the inner periphery of the IU could, under certain conditions, maintain the astrionic equipment within their temperature tolerances, or at least minimize the need for additional heat rejection or heat addition. Furthermore, the expendable heat sink — water and sublimator - is perhaps the only practical means for heat rejection during the initial powered flight period of the mission. With modifications, which would include replacing components or parts that have limited design life and eliminating all or part of the dependency on expendable supplies, the current system could be used for extended periods.

Using the current design concept as the baseline, various modified baseline concepts have been synthesized, ranging from the minimum modified system for the short missions to the more extensive modified system for the long missions. These concepts represent the major or basic features required to meet the functional requirements of the TCS.

BASELINE ECS

The current ECS has been reviewed to determine the significant design limitations for extending the operational time beyond the present six and one-half hours. A schematic of the current system is given in Figure 2-1, which illustrates the thermal conditioning and gaseous supply subsystems. This is a semi-closed system that is heavily dependent upon stored or expendable supplies.

Component failure and the consumption of the expendable supplies are general features that limit the operational time. With regard to component failure, components that have rotating and/or moving parts, such as the centrifugal pump and the modulating flow control valve, can be assumed to be designed to the specific operating time limit and, thus, must be replaced or redesigned for the longer time period. Components that have no moving parts could be used indefinitely, with the exception of the coldplates, which have a considerable area (100 square feet) that is vulnerable to meteoroid damage or puncture. The longer mission time and the possible exposure of the forward end of the IU to space environment increase the probability of such failure. The present coldplates could be used for the extended period if they were adequately protected.


SUPPLY
Orifice Assembly
Coolant Temperature Sensor
Redundant Coolant Pumps
Cold Plate (16 Req'd)
Methanol-Water Accumulator
Flexible Membrane
Thermal Expansion Chamber
Umbilical
Preflight Heat Exchanger
GSE Supplied Coolant
In
Out
Flight Heat Exchanger (Sublimator)
Vapor Vent
Water Solenoid Valve C
Water Fill Port
Water Accumulator
Flexible Membrane
Modulating Flow Control Valve
Bypass
Electronic Controller Assembly
GN2 Vent
Orifice Regulator
Calibrate
GN2 Supply & Emergency Vent
Shut-off Valve
Quick Disconnect (Typ)
SUPPLY
Cold Plate (16 Req'd)
Legend
Methanol-Water Coolant
Methanol-Water (GSE Supplied)
Gaseous Nitrogen (GN2)
Water and Water Vapor
Calibrate Line (Removed Prior to Flight)

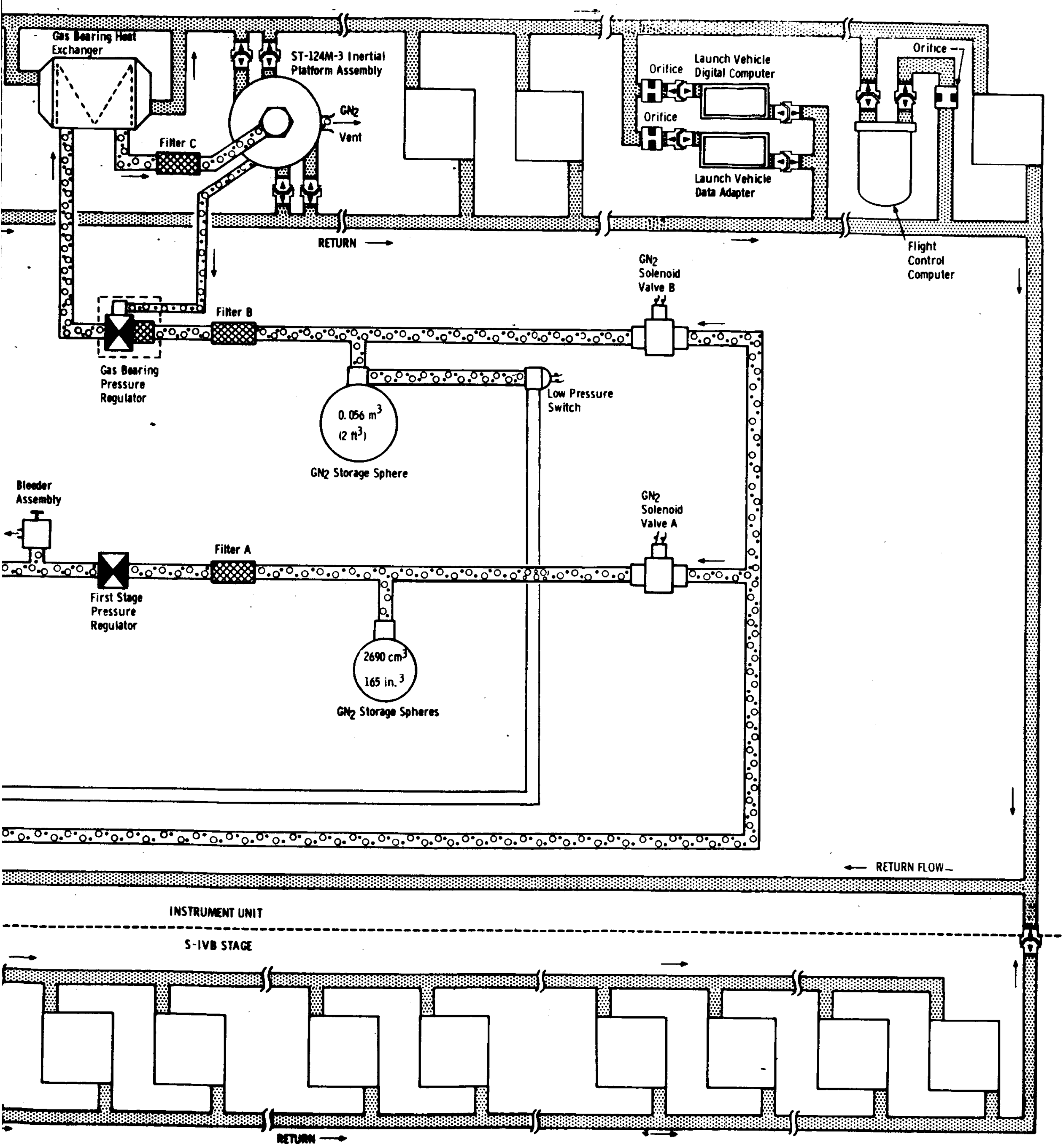


Figure 2-1. Environmental Control System Simplified Schematic

The second significant feature that limits the operational time of the current ECS is the expendable supply, which includes the methanol-water coolant, gaseous nitrogen, and cooling water (see Reference 2-1). Based on the capacity of the methanol-water accumulator and an assumed leakage rate of 0.7 cubic inch per hour, the lifetime of the coolant circuit is limited to 130 hours. However, on the basis of the volume of GN_2 required for methanol-water leakage makeup, the lifetime of the current ECS is only about 15 hours, and the volume of GN_2 available for the ST-124M inertial platform gas-bearing system is sufficient for only 12 hours of operation under the specification flow rate of 0.5 standard cubic foot per minute. The limit on the lifetime of the ECS imposed by the available cooling-water supply (148 pounds) varies with orbital condition and the level of electronic equipment power dissipation. For the orbits and vehicle orientations under consideration for this study, the lifetime limits imposed by the cooling-water supply are listed in Table 2-1 for a coolant temperature of 50 F and two equipment power dissipation rates. The first of these rates, 3.0 kilowatts, represents the lower limit for equipment applicable to the 1966-1969 operational time period, as indicated in Figure 1-5 under orbital checkout and orbit full-on conditions. The other power dissipation rate, 3.9 kilowatts, represents the total for the equipment on the present IU. It may be seen that, for the conditions selected, the minimum lifetime of the cooling-water supply system is 41 hours. The last two columns in Table 2-1 show the coolant temperatures required in order that no cooling water is necessary for temperature control. If these coolant temperatures were to be acceptable, the lifetime of the cooling-water supply system would not be a limiting factor. Thus, it appears possible that the current system could be used beyond the six and one-half hours to about 12 hours with no design modifications. For the longer periods, modifications such as replacing certain parts, increasing the amount of expendables, and adding new components are necessary.

MODIFIED BASELINE CONCEPTS

Starting with the current design concept, minimum modifications for extending the operational time were considered first, and progressively more extensive changes or modifications were considered, to satisfy the longer mission time and the other system requirements. The underlying considerations for any modifications of the baseline system were weight and reliability, which have been assumed to be the most significant criteria. The limitations of the present system that were discussed in the previous paragraph were based on these two criteria. Other considerations were the possible design improvements of the present astrionic equipment and the future trend of the design and development of electronic equipment. These considerations, in addition to the requirements and constraints described in Section 1.0, provide the basis for the synthesis of the modified baseline concepts.

Table 2-1. Lifetime of Expendable Water System on Present IU Thermal Conditioning System

Orbit Designation	Altitude (n. mi.)	Angle of Inclination (deg)	Launch Date	Spacecraft Orientation	Cooling Water Supply Lifetime (hr) at 50 F Coolant Temp		Coolant Temperature (F) for Zero Water Requirement	
					Q_{el} = 3.0 kw	Q_{el} = 3.9 kw	Q_{el} = 3.0 kw	Q_{el} = 3.9 kw
IU-1b-2	200	29	June 21	X axis sun-oriented	296	67	54	68
IU-1c-3	200	29	June 21	Y axis sun-oriented	99	48	63	76
IU-3a-4	200	90	March 21	X axis along orbit path	74	41	67	81
IU-3b-5	200	90	March 21	X axis sun-oriented	296	67	54	68
IU-3c-6	200	90	March 21	Y axis sun-oriented	74	41	67	81
IU-4a-7	100	0	March 21	X axis along orbit path	99	48	63	76
IU-6a-8	19,327	0	March 21	Y axis sun-oriented	296	67	54	68
IU-6b-9	19,327	0	March 21	X axis sun-oriented	*	296	40	56

Q_{el} = Equipment power dissipation
*Requires heat addition to coolant

For the initial concepts, no changes in the basic design or locations of the thermal conditioning panels were assumed, since these panels represent the major portion of the current system design, and any basic changes would constitute a major modification. Any changes to the other components because of potential component failure have been assumed to be minor and could be accomplished readily without an appreciable increase in system weight. Various approaches to the stored supplies (or expendables) have been considered, including increasing the amount of expendables, reducing the usage rate, or eliminating all or part of the requirements for expendables. Based on these considerations, four concepts have been synthesized, representing progressive increase in the degree of modification required to the baseline concept, as well as increased integration with other subsystems to meet the long-duration operational time with minimum increase in system weight. It was assumed for these initial concepts that the S-IVB electronic equipment would be utilized only for the control of the S-IVB stage and, thus, would not be used after restart to change from the checkout orbit to the final mission orbit. Thus, shutoff valves programmed to shut off automatically after the S-IVB stage has been completely spent would be added to the lines to and from the S-IVB coolant loop.

Modified Concept 1

The simplest minimum modification to extend the operating time beyond six and one-half hours would be to replace the components that have limited design life and to increase the stored quantity of water and gaseous nitrogen for the ST-124M bearings by adding storage tanks. The amount of increase in the stored water and gaseous nitrogen would be limited by mission and vehicle constraints, since the weight and volume of the stores could reach prohibitive values. This situation is shown in Table 2-2 in which required weights or volumes of expendables as a function of mission duration are listed. Quantities available in the present system are shown for the purpose of comparison. These latter quantities and the leakage rates used to calculate required amounts were obtained from Reference 1-2. It is apparent that the operating time of the present system is limited only by the amount of permissible increase in stored expendables.

In addition, changes in the coolant flow rate and surface coatings of the thermal conditioning panels and surrounding structure may be required to utilize to the fullest the thermal conditioning panels for heat rejection or heat retention. Another change would be to use a smaller orifice size for the pressure regulation of the cooling-water storage tank.

Modified Concept 2

This concept is similar to Concept 1 except that the stored gaseous nitrogen for the ST-124M gas bearing is replaced by a gaseous nitrogen recirculating system, and, in addition to the stored water, expendables

Table 2-2. Weight of Required Expendables Versus Mission Duration

Mission Duration	GN_2 for ST-124M Gas Bearing System (lb)	GN_2 for Methanol/Water Accumulator (lb)	Methanol/Water Leakage (in.^{3})	Cooling Water for 3-kw Heat Load (lb)
10 hours	21.75	0.90	7.0	100
48 hours	104	4.32	33.6	480
10 days	544	21.6	168	2,400
30 days	1,630	64.8	504	7,200
90 days	4,890	195	1510	21,600
180 days	9,780	389	3,020	43,200
Capacity of present system	28	1.35	90	148

from other sources would be utilized. One alternative, which depends upon the use of a fuel cell, would be to use the water from the fuel cell to augment the stored water supply. Only the addition of a water accumulator would be required. A second possible alternative would be to utilize the hydrogen vent gas from the S-IVB stage, in the event such gas is available as a heat sink material during orbital coast. After final burn of the stage, the remaining quantity of hydrogen (173-701 pounds, according to Reference 2-2) may be available if utilization of the stage does not call for immediate dumping of the hydrogen. Use of cryogenic hydrogen gas would require the addition of a hydrogen-methanol cooler in the thermal conditioning loop.

Concept 2 is a step toward limiting the amount of stored supplies and, thus, minimizing any weight increase for extending the operating time.

Modified Concept 3

This concept is similar to Concept 2 except that a space radiator would be added to the thermal conditioning loop for heat rejection, and expendables from other sources would not be used. The stored water would be used for cooling during the ascent period and on a demand basis for the rest of the mission (primarily for periods of peak heat loads). An alternative would be the case in which the radiator is designed to reject all the heat during the orbital phase so that stored water would not be required. A radiator designed to meet both the peak and sustained heat loads could present radiator freezing problems.

Concept 3 introduces a new interface with IU structure or other adjacent structure for which modifications would be required.

Modified Concept 4

This concept is similar to Concept 3 except that it would depend on water from fuel cells to augment the stored water supply. In this concept, the radiator would be designed for the steady heat load, and the stored water plus the fuel cell water would be used for the peak heat load periods.

In these initial modified concepts, no change in the configuration of the thermal conditioning loop was assumed, thus minimizing the extent of modifications. The next step was to consider various modifications to the thermal loop for better utilization and possible reduction in the heat loads to the thermal loop and the associated heat rejection device.

A review of the current astrionic equipment that is coldplate-mounted (presented in Section 1.0) has indicated that, on the basis of the allowable case temperature tolerance, the equipment could be placed into three groups: (1) Group I, with the narrowest temperature tolerance range (50 F to 122 F), is assumed to require an active ECS, which would consist of a recirculating coolant loop with a means for heat rejection, in addition to the thermal conditioning panels. (2) Group II, with a greater temperature tolerance range (-4 F to 167 F), is assumed to require only a recirculating coolant loop, which would depend upon the thermal conditioning panels for heat rejection. By separating Group I and Group II, the heat load to the heat rejection device, such as the expendable heat sink or space radiator, would be reduced. The Group I thermal loop and the Group II thermal loop could be connected thermally by a common heat exchanger with a by-pass that would permit heat transfer from one loop to the other or no heat transfer, depending upon thermal requirements. (3) Group III, with the widest temperature tolerance range (-67 F to 185 F), is assumed to require only a passive means for thermal control. There are two possible alternatives for mounting the equipment. One method would be to mount the equipment directly to the IU inner structure so that the equipment heat would be transferred by the combination of conduction and radiation. Although this is a simple approach, it may not be feasible because of the ascent heating. The other alternative would be to mount the equipment on the thermal conditioning panel and provide a thermally controlled shutoff valve at the coolant outlet of the panel. The panel(s) would be located in the Group II thermal loop so that, whenever passive cooling is not sufficient, the thermally controlled valve would allow the coolant to flow and remove the excess heat.

This approach of grouping and localizing the heat rejection is another means for reducing the heat rejection load on an expendable heat sink or space radiator, which could result in weight savings. However, any benefit would be partially offset by the increased complexity of the overall system.

Table 2-3 summarizes the alternative approaches for thermal control on the basis of astrionic equipment grouping and mission periods. The TCS concepts to be selected will be total concepts, encompassing the three equipment groups and the various mission periods on an integral basis.

As indicated in Section 1.0, a review of future astrionic equipment design points to a trend toward integrally cooled equipment rather than packages which transfer power dissipation heat by conduction to a coldplate. The review also reveals trends toward lower package heat loads, little change in package temperature tolerance, and smaller package size. These trends suggest a reduction of the number of thermal conditioning panels required for equipment cooling. For this situation, one approach would be to maintain the present thermal conditioning configuration with no change in the number of thermal conditioning panels. These panels would be used for equipment mounting, including those that are integrally cooled (flight control computer, launch vehicle digital computer, launch vehicle data adapter, and ST-124 inertial platform assembly), and panels that are not used for equipment heat sink would be utilized for heat rejection and in conjunction with other heat rejection devices. Each of these panels would be equipped with a thermally controlled shutoff valve at the outlet from the panel to control the heat rejection. In some instances, some of the panels could serve as a heat source, by absorbing the environmental heat, to maintain the desired overall heat balance.

An alternative approach would be to remove the panels that are not required and to rearrange the panels for maximum utilization as both heat sinks and heat rejection devices. In addition to the expendable heat sink, a space radiator would be used for the necessary heat rejection. The advantage of this approach would be the more efficient overall heat rejection capability since the space radiator is more efficient for heat rejection than the thermal conditioning panels.

On the basis of the projected design changes indicated in Table 1-3 for the future operation equipment, the present 16 thermal conditioning panels could be reduced to about eight panels required for electronic packages that would require coldplates for heat sink. The other equipment would be

Table 2-3. Alternative Approaches for Thermal Control

Equipment	Thermal Control Approaches for Mission Periods		
	Prelaunch	Initial Powered Flight	Orbit
Group I	GSE HX	Active: Sublimator	Active: Sublimator and/or radiators, plus coldplates as secondary radiators
Group II	GSE HX	Closed liquid loop (Active: sublimator)	Closed liquid loop with coldplates as secondary radiators (radiation only)
	GSE HX	Closed liquid loop with coldplates as secondary radiators (radiation only); S-IVB dome used as heat sink	(No cooling by sublimator or radiator) (Coolant circulating through coldplates)
Group III	Combined with Group II TCS	Combined with Group II TCS	Passive (radiation only); Coldplate mounted
	Passive; IU skin mounted	Passive; IU skin mounted	Passive (radiation only); IU skin mounted

integrally or internally thermal-conditioned. In addition to weight reduction by reducing the number of thermal conditioning panels, the operating conditions may change—such as a wider temperature range for the coolant—which could further minimize the need for heat rejection by expendable means or reduce the space radiator size.

These possible changes to the thermal conditioned panels and, hence, to the system configuration, can be combined with the four previous concepts in various combinations to arrive at additonal concepts that represent a greater degree of modification.

3.0 SYSTEM ANALYSIS

To aid in establishing the conditions under which the possible modifications to the baseline thermal conditioning system will apply, a comprehensive analysis of the TCS was made. The purpose of the analysis was to determine the influence of a number of system variables, such as coolant flow rate and thermal control coating properties, on system performance and requirements under several different orbit conditions. Results of the analysis were used to determine the modifications of the baseline concept needed to extend its operating time. The analyses that were performed are summarized in the following paragraphs.

FLUID TEMPERATURE LIMITS

The tolerable temperature ranges for the methanol water coolant, or other organic heat transfer fluids (e.g., 1 butene) in the coldplates and in the integrally cooled equipment, were established for the equipment now in the IU. This was done to establish the maximum tolerable range and the highest permissible upper temperature and to identify the electronic packages that restrict or limit the coolant temperature range on an overall system basis. The influence of coolant flow rate, reduction in joint thermal resistance, and modification in the equipment case temperature tolerances for NASA specifications have been established.

Figure 3-1 shows the upper fluid temperature limits, based on NASA equipment specification case temperature limits (References 1-1, 1-4, and 1-5). It is apparent that the ST-124M inertial platform assembly is the critical package and that it limits the upper fluid temperature to about 80 F. The next upper limit is about 85 F, imposed by the launch vehicle data adapter. Figure 3-2 shows the upper temperature limits on the basis of modified case temperature limits. Comparison of Figures 3-1 and 3-2 shows the benefit of the suggested modified case temperature limits in raising the upper fluid temperature limit for a number of the electronic packages. However, the launch vehicle data adapter still imposes an upper limit of about 85 F.

Figure 3-3 shows the lower fluid temperature limits for both NASA specification and NAA-modified case temperature limits. The figure indicates that the accelerometer signal conditioner is the critical package, limiting the lower fluid temperature to about 59 F. Based on NAA-modified case temperature limits, the lower fluid temperature limit would be about 49 F and would be imposed by the same package.

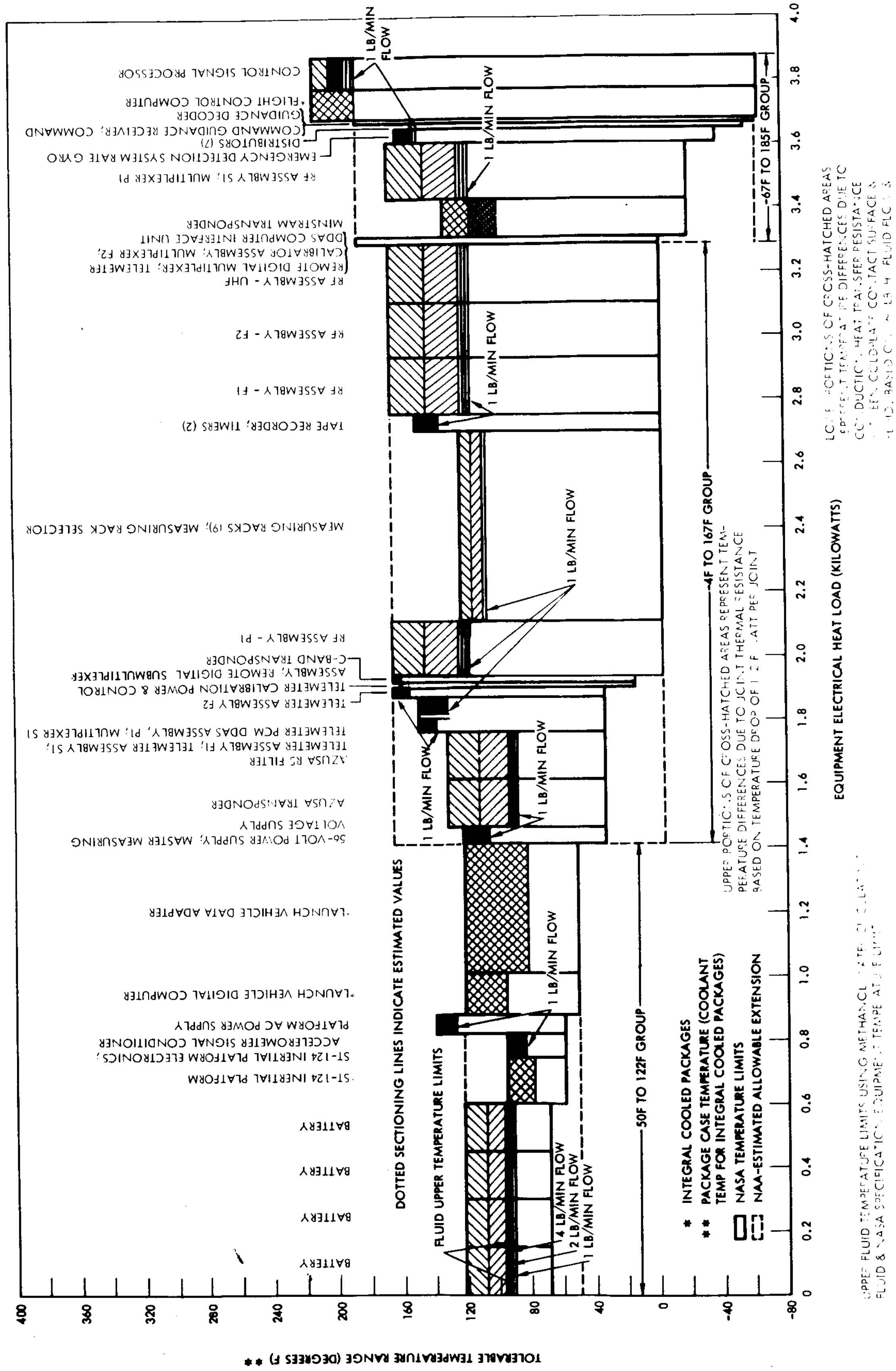


Figure 3-1. Upper Fluid Temperature Limits, NASA Specifications Equipment Temperature Limits

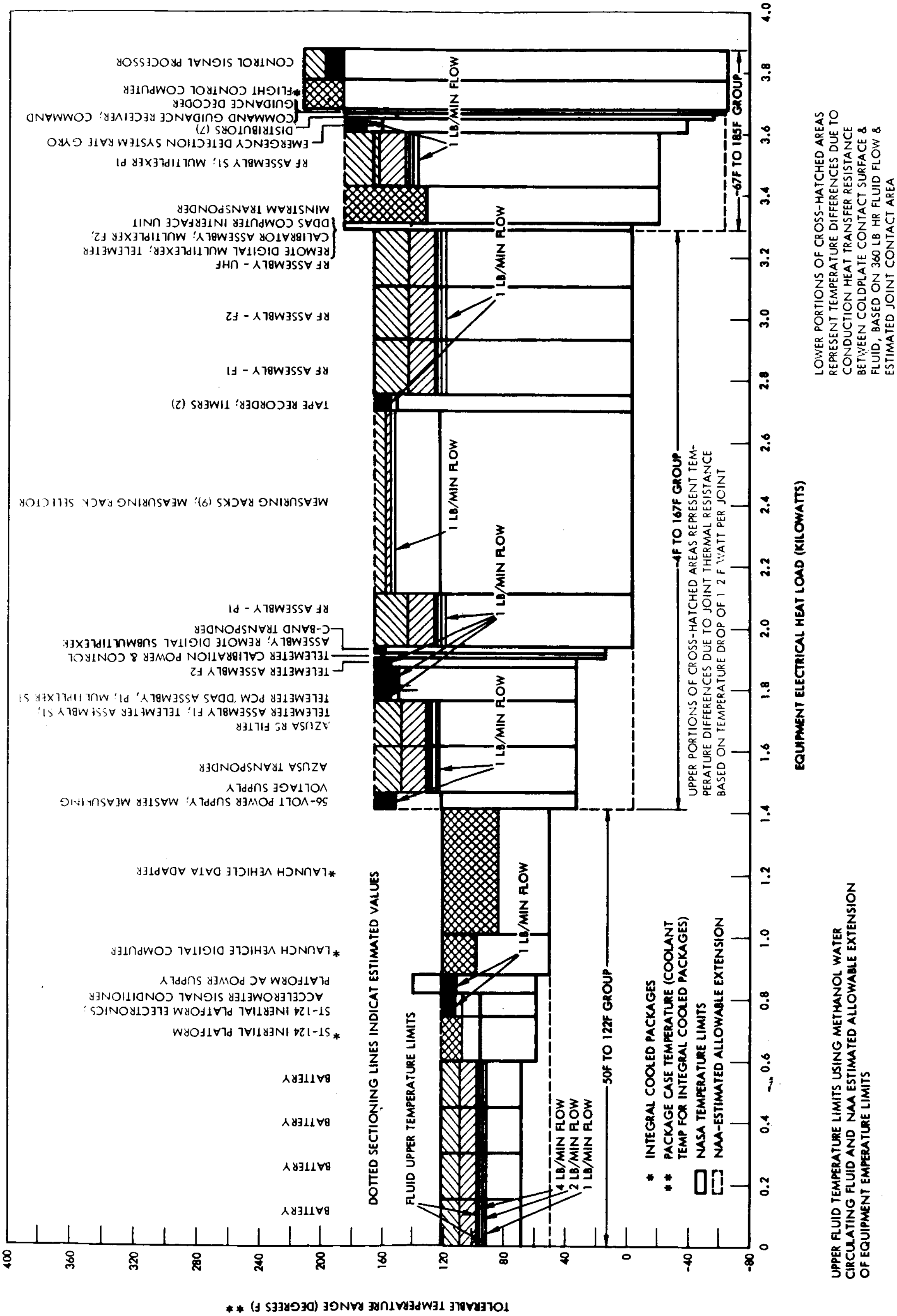


Figure 3-2. Upper Fluid Temperature Limits, NAA Modified Temperature Limits

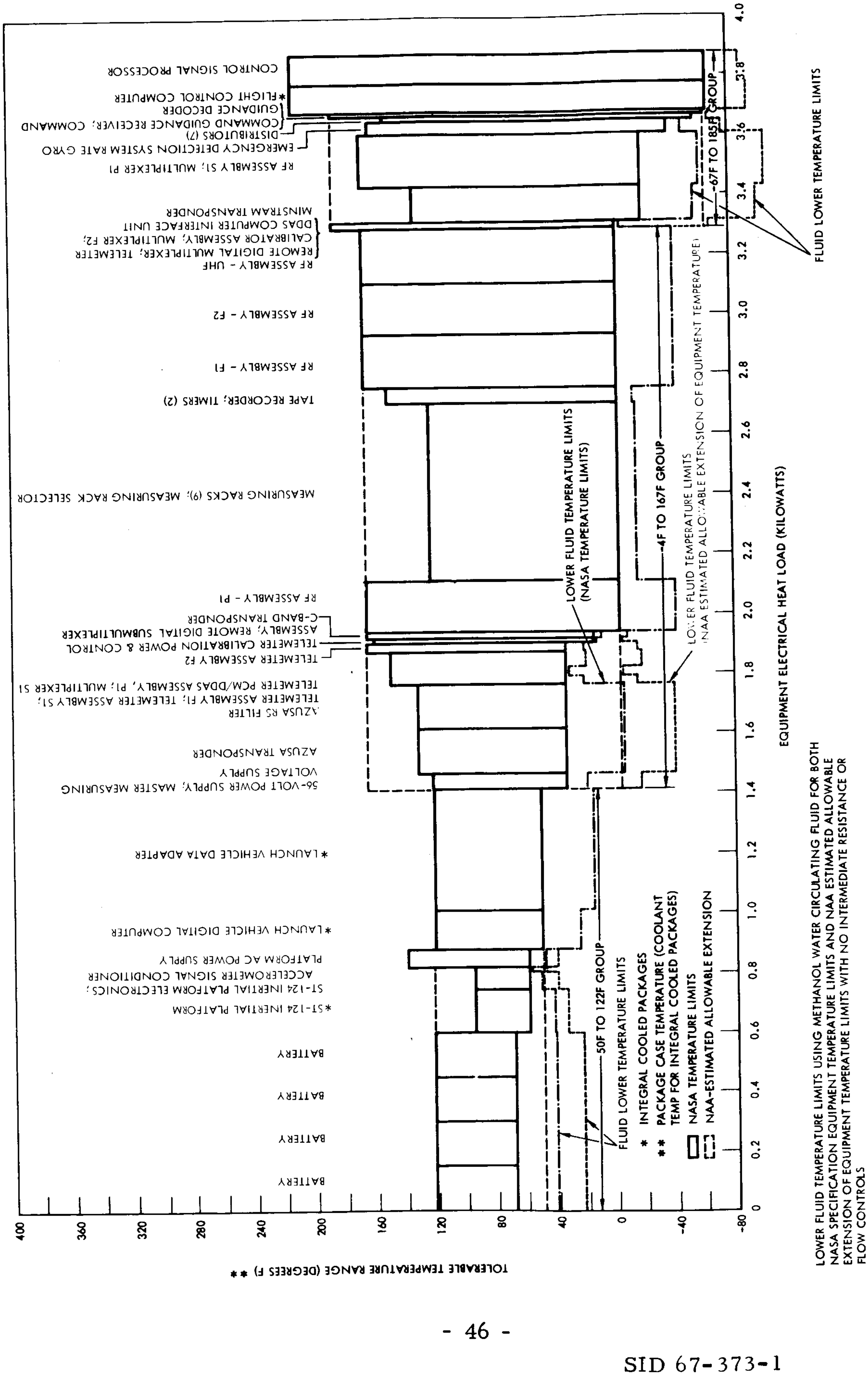


Figure 3-3. Lower Fluid Temperature Limits (Based on NASA Specification Equipment Temperature Limits and NAA Modified Equipment Temperature Limits)

Figure 3-4 shows the influence on the fluid upper temperature limit of reducing the joint thermal resistance by using dead soft aluminum.

THERMAL ANALYSES (HEAT BALANCE)

Major portions of the system analysis consisted of determining the heat balances for the current, or baseline, thermal conditioning system, consisting of a closed liquid coolant loop with coldplates and integrally cooled electronic packages. These heat balances were calculated by using two digital computer programs, one to establish the environmental heat flux and the other a general heat transfer program for the thermal analysis.

Among the parameters required by the orbital heating program as input data are orbital altitude, angle of inclination of the orbit, launch date, and spacecraft orientation. These parameters are listed in Table 3-1 for the various cases investigated. In connection with these data, it is pointed out that only circular orbits were considered. The spacecraft orientation is based on specifying the direction of spacecraft surface normals with respect to an orthogonal coordinate system, which is itself related to the appropriate orbit coordinate system. The X axis of the orthogonal system was chosen to coincide with the longitudinal axis through the center of the S-IVB IU combination.

Table 3-1. Orbital Parameters

Case No.	Altitude (nautical miles)	Angle of Inclination (degrees)	Launch Date	Spacecraft Orientation
IU-1a-1 IU-1b-2 IU-1c-3	200	29	June 21	X axis along orbit path X axis toward sun Y axis toward sun
IU-3a-4 IU-3b-5 IU-3c-6	200	90	March 21	X axis along orbit path X axis toward sun Y axis toward sun
IU-4a-7	100	0	March 21	X axis along orbit path
IU-6a-8 IU-6b-9 IU-6c-10	19, 327 (synchronous)	0	March 21	Y axis toward sun X axis toward sun X axis along orbit path

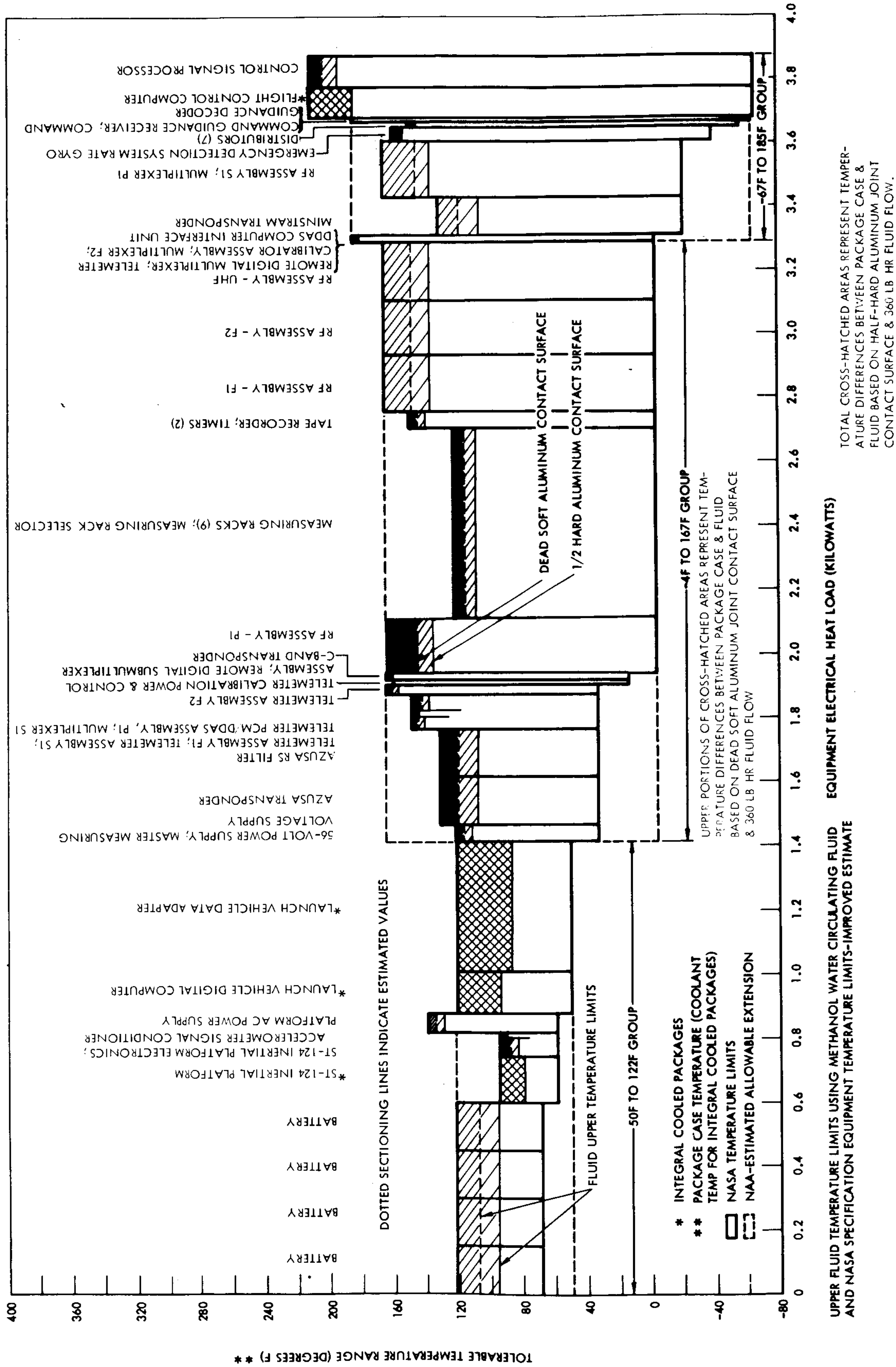


Figure 3-4. Upper Fluid Temperature Limits, Effect of Reduced Joint Thermal Resistance

Table 3-1 shows that only one vehicle orientation was investigated for the 100-nautical-mile orbit while three vehicle orientations were used for each of the other three orbits. This was done because a limit had to be placed on the scope of the investigation, and the 200-nautical-mile and synchronous earth orbits were considered to be more appropriate for future missions than the 100-nautical-mile orbit. In addition, the determination of incident thermal radiation indicated that there is very little difference between the amounts of such radiation at altitudes of 100 and 200 nautical miles.

Other factors that determine the magnitude of the incident heat load are the solar constant, magnitude of albedo radiation, and resulting earth mission. Values for these factors, as used in the orbital heating program, are 443 Btu/hr-ft^2 for the solar constant, 40 percent of solar radiation for albedo radiation, and 66.4 Btu/hr-ft^2 for earth-emitted radiation. These values are the same as those used in Reference 3-1. The variation in the value of the solar constant with variation in launch date was not considered significant and, therefore, was not taken into account.

The vehicle configuration selected for the thermal analysis consists of an IU, an S-IVB stage aft of the IU, and a spacecraft/LM adapter (SLA) forward of the IU. For the purpose of the computer analysis, the complete IU was represented by an equivalent electrical network. The IU was divided into 24 equal segments corresponding to the present 24 IU locations.

For the purpose of the thermal analysis, it was assumed that coldplate-mounted equipment would cover one half of the available area (30 by 30 inches) of each coldplate. Hence, radiating coldplate and electronic package areas were considered as 450 square inches each. The electronic package weight assigned to each coldplate location was determined by dividing the total electronic package weight for coldplate-mounted equipment (2370.3 pounds) by the number of coldplates (16). This approach results in a thermally ideal system, as compared with the package arrangement on the present IU, which is on a functional basis. However, the equal distribution of electronic package weight and, hence, heat capacity was not expected to affect the total heat load on the cooling system, and the rate of change in the incident heat load was expected to be slow enough to make the effect of heat capacity on individual package temperatures negligible. This expectation was reinforced by the assumption, for analysis purposes only, of constant coldplate coolant inlet temperature, regardless of the incident heat load on the IU outer shell.

Further assumptions applicable to the construction of the IU equivalent network and the analysis of the network are as follows:

1. The single node representation for each coldplate is sufficiently accurate for obtaining data to be used in a TCS selection.

2. The heat capacity of the coolant within the coldplate passages is small enough to be neglected in the equivalent network representation.

3. The temperature difference between the IU and adjacent structure is small enough to neglect conduction heat transfer between the two.

4. The electronic package cover and the coldplate surface are equivalent to a single flat surface for the purpose of defining internal thermal radiation. In the absence of detailed information regarding package dimensions and relative position on the coldplates, this assumption is believed to produce results of reasonable accuracy because the omission of radiation from the package sides to the S-IVB dome and the SLA is offset by neglecting the shadowing effect of the packages with respect to radiation from the bare coldplate surface area.

As indicated in Table 3-2, the electronic package heat load for coldplate-mounted equipment was also assumed to be distributed uniformly; and, as mentioned in the previous discussion of heat capacity, this thermally ideal distribution was not expected to affect total system heat load. Heat loads for integrally cooled items of equipment are, of course, actual loads.

In this connection, it should be noted also that, for the earth-oriented orbital conditions (Cases 1, 4, 7, and 10 in Table 3-1), only one value of electronic package heat load (150 watts per coldplate) was used in the heat balance calculations. For the other six orbital cases investigated, heat balances were obtained over the electronic package heat load range from 50 to 250 watts per coldplate. The reason for limiting the number of heat balances obtained for Cases 1, 4, 7, and 10 is that IU heat load values in these cases will fall between heat loads for the other two vehicles orientations (X axis toward sun and Y axis toward sun), which represent minimum and maximum solar heating conditions for any one orbital condition. The values for other variables affecting TCS requirements, such as coolant coldplate inlet temperatures, IU outer shell solar absorptivity, and coolant flow rate, are listed in Table 3-2.

The value of the conductors representing radiation heat transfer paths was determined on the basis of the surface emissivities listed in Table 3-2. For the S-IVB dome, a possible range of emissivity value between 0.05 and 0.90 was assumed. It was considered desirable to enhance radiation heat transfer from the electronic packages, and so they were assumed to be covered with a thermal control coating of high emissivity ($\epsilon = 0.9$). The same emissivity value was assumed for the outboard surface of the IU outer

Table 3-2. Conditions Used in Thermal Analysis

ELECTRICAL HEAT LOADS	
Coldplates	50, 150, and 250 watts per coldplate
ST-124M inertial platform assembly	70 watts
LVDC	142 watts
LVDA	400 watts
Flight control computer	100 watts
ECS VARIABLES	
Coldplate coolant inlet temperature	30, 50, and 75 F
IU skin solar absorptivity (α_s)	0.18 and 0.9
Coolant flow rate (per coldplate)	60 lb/hr
SURFACE INFRARED EMISSIVITIES	
S-IVB dome	0.05 and 0.90
SLA (spacecraft/LM adapter)	0.18
Coldplates	0.18
Electronic packages	0.90
IU skin:	
Outboard surface	0.90
Inboard surface	0.18

shell, which must be effective in rejecting absorbed incident radiation to the deep space heat sink. For the rest of radiating surfaces (coldplates, SLA, and the inboard surface of the IU outer shell), an emissivity value corresponding to the lowest available with commonly used thermal control coatings was used. Table 3-3 gives the values for conductance and fA's used in the radiation heat transfer.

The starting time for the thermal analysis was considered to be that of injection into orbit, and the IU skin temperature at that instant was assumed to be 200 F. The corresponding S-IVB dome temperature was assumed to be -210 F. Since the S-IVB dome temperature does not reach equilibrium conditions until approximately 30 hours after launch, heat balances were calculated for a total of 40 hours to evaluate the effect of variations in this temperature.

Figures 3-5 through 3-11, showing the relationship between coolant temperature at coldplate inlet and equipment total power dissipation, were derived from the results of the thermal analysis. These figures were derived from the data presented in the appendix to this volume of the report. Table 3-4 lists the case number and the figures in the appendix from which they were derived.

Table 3-3. Values for Conductance and fA's

Component	Component/IU Skin		Component/S-IVB Dome fA (sq in.)		Component/SLA fA (sq in.)	
	Radiation	Conduction				
	fA (sq in.)	$\frac{kL}{A}$ (Btu/(hr) (F)	S-IVB Dome $\epsilon = 0.05$	S-IVB Dome $\epsilon = 0.90$	S-IVB Dome $\epsilon = 0.05$	S-IVB Dome $\epsilon = 0.90$
Coldplate	104	14.28	10.0	91.5	36.0	14.0
Electronic package (coldplate mounted)	-	-	28.0	267.0	105.0	40.0
ST-124 inertial platform	173	-	95.0	902.0	356.0	136.0
LVDA	198	-	159.0	1505.0	595.0	228.0
LVDC	185	-	126.0	1195.0	473.0	181.0
Flight control computer	202	-	174.0	1650.0	652.0	250.0

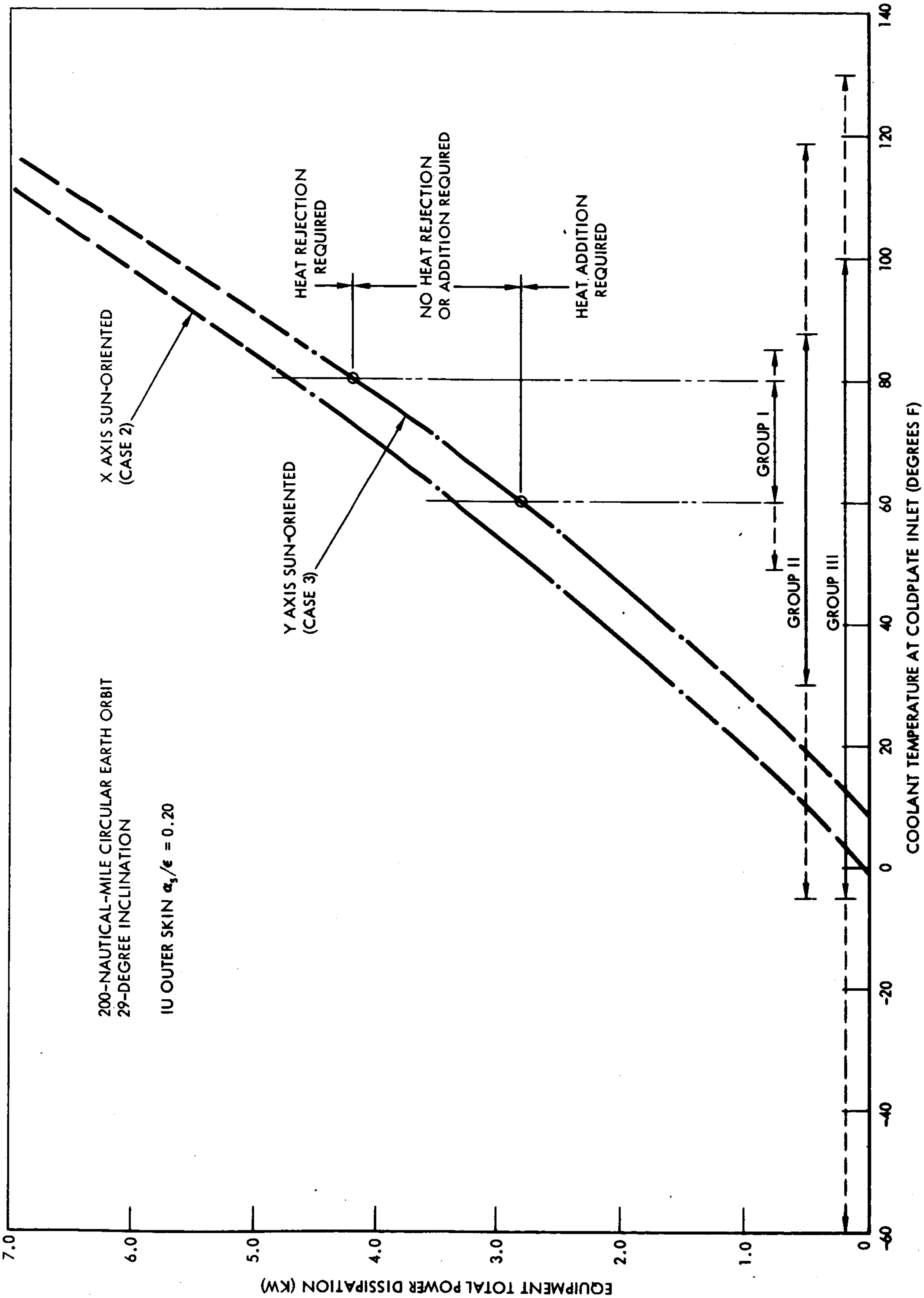


Figure 3-5. Coolant Temperature Versus Equipment Acceptable Total Power Dissipation

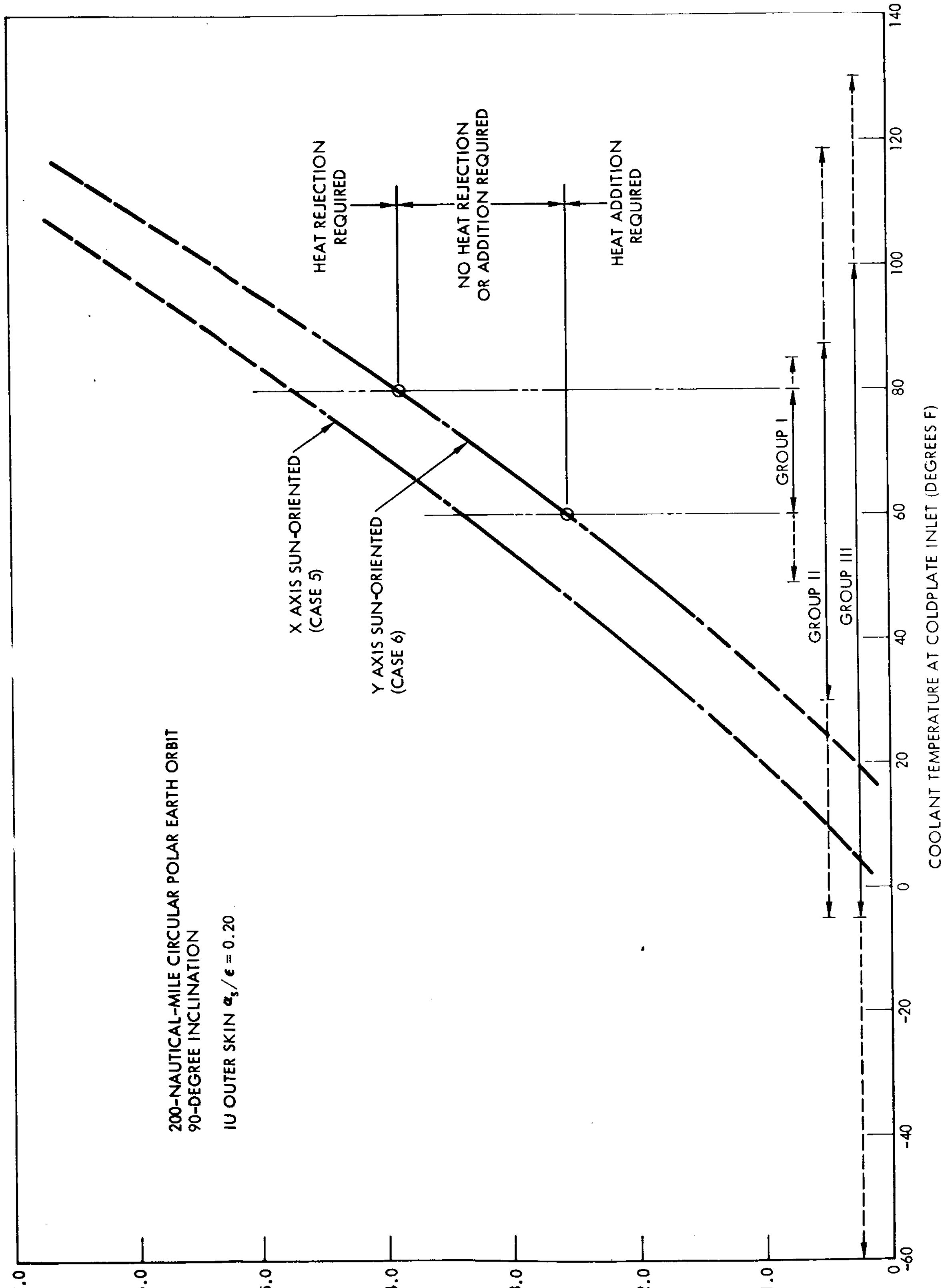


Figure 3-6. Coolant Temperature Versus Equipment Acceptable Total Power Dissipation

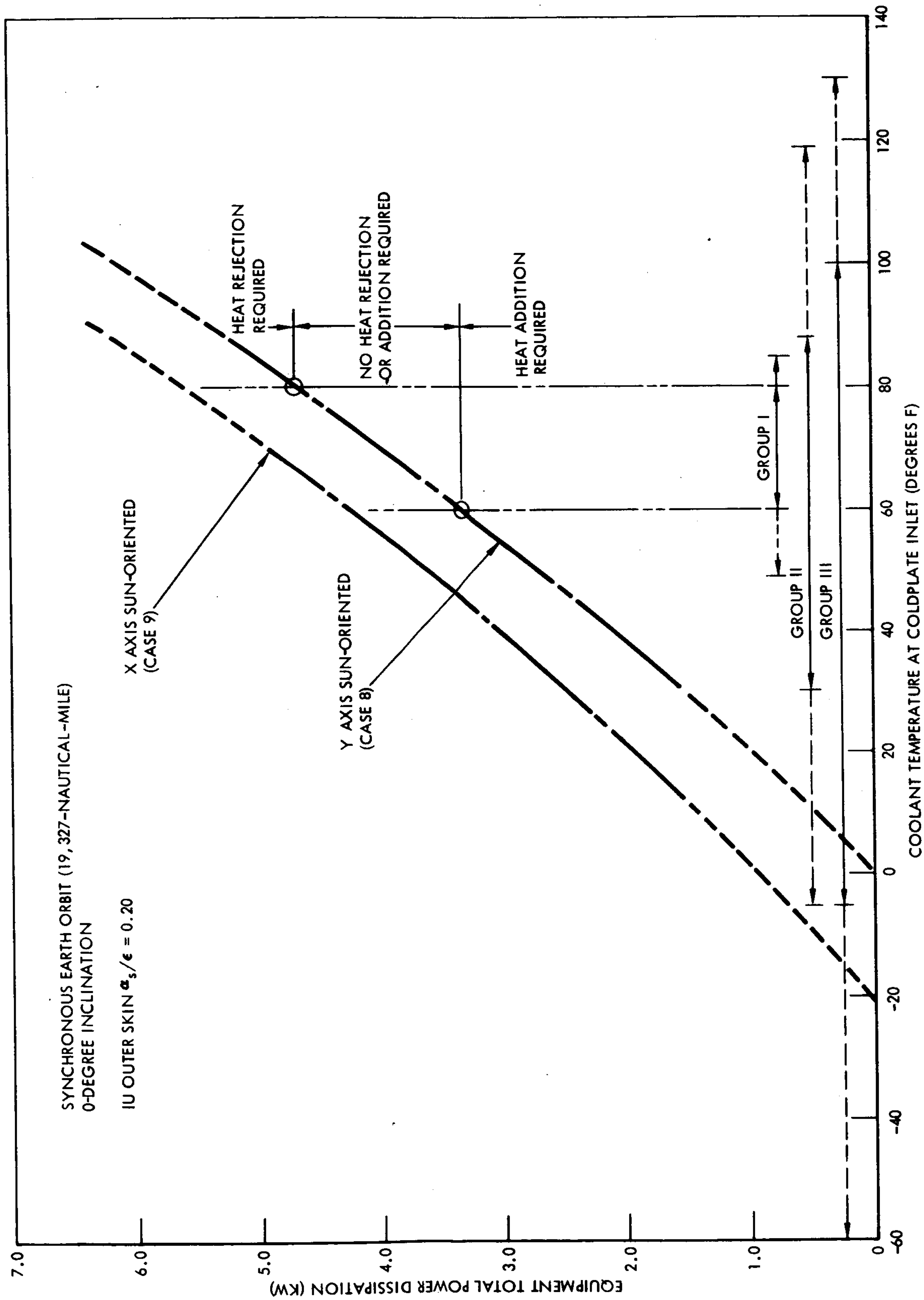


Figure 3-7. Coolant Temperature Versus Equipment Acceptable Total Power Dissipation

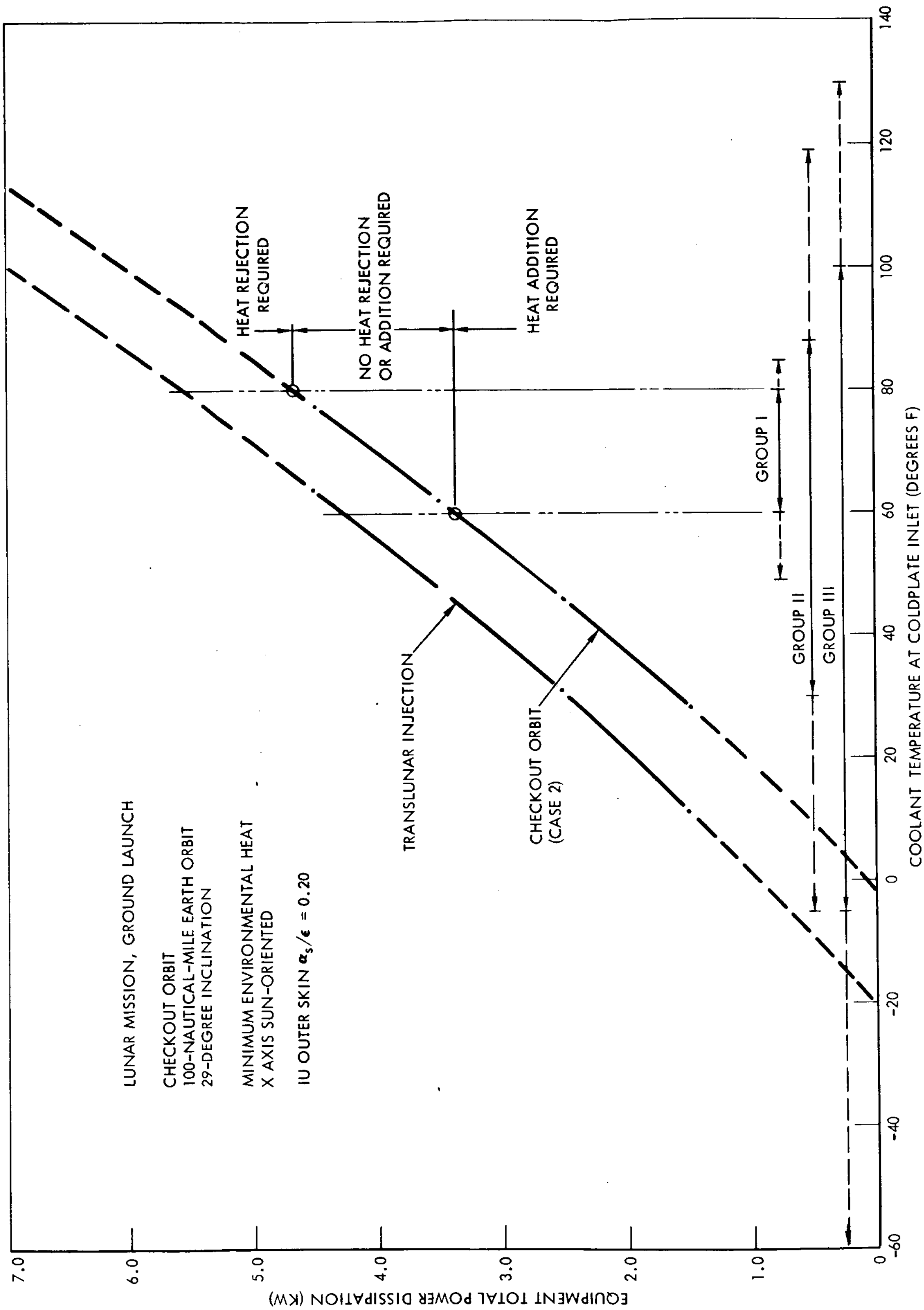


Figure 3-8. Coolant Temperature Versus Equipment Acceptable Total Power Dissipation

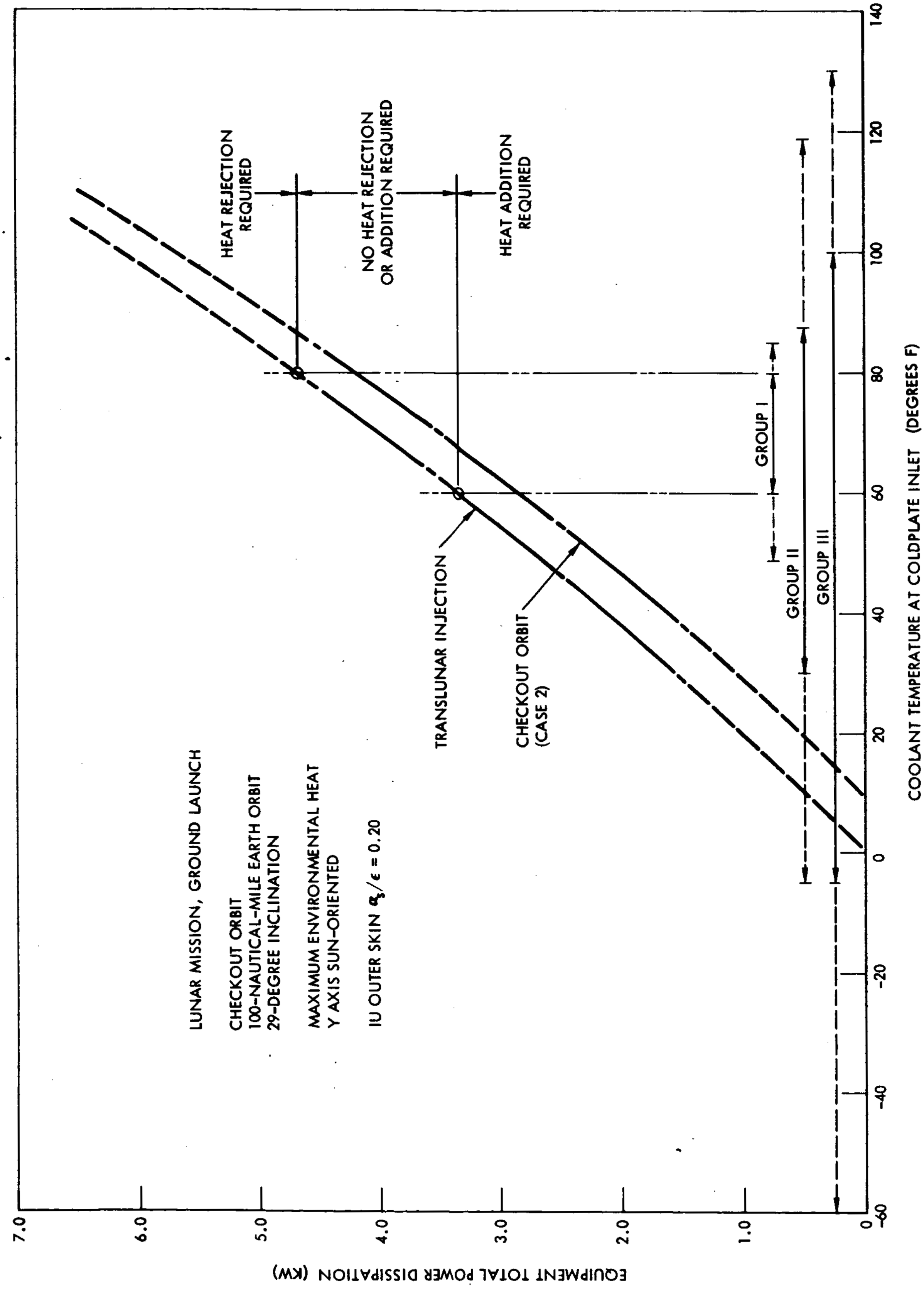


Figure 3-9. Coolant Temperature Versus Equipment Acceptable Total Power Dissipation

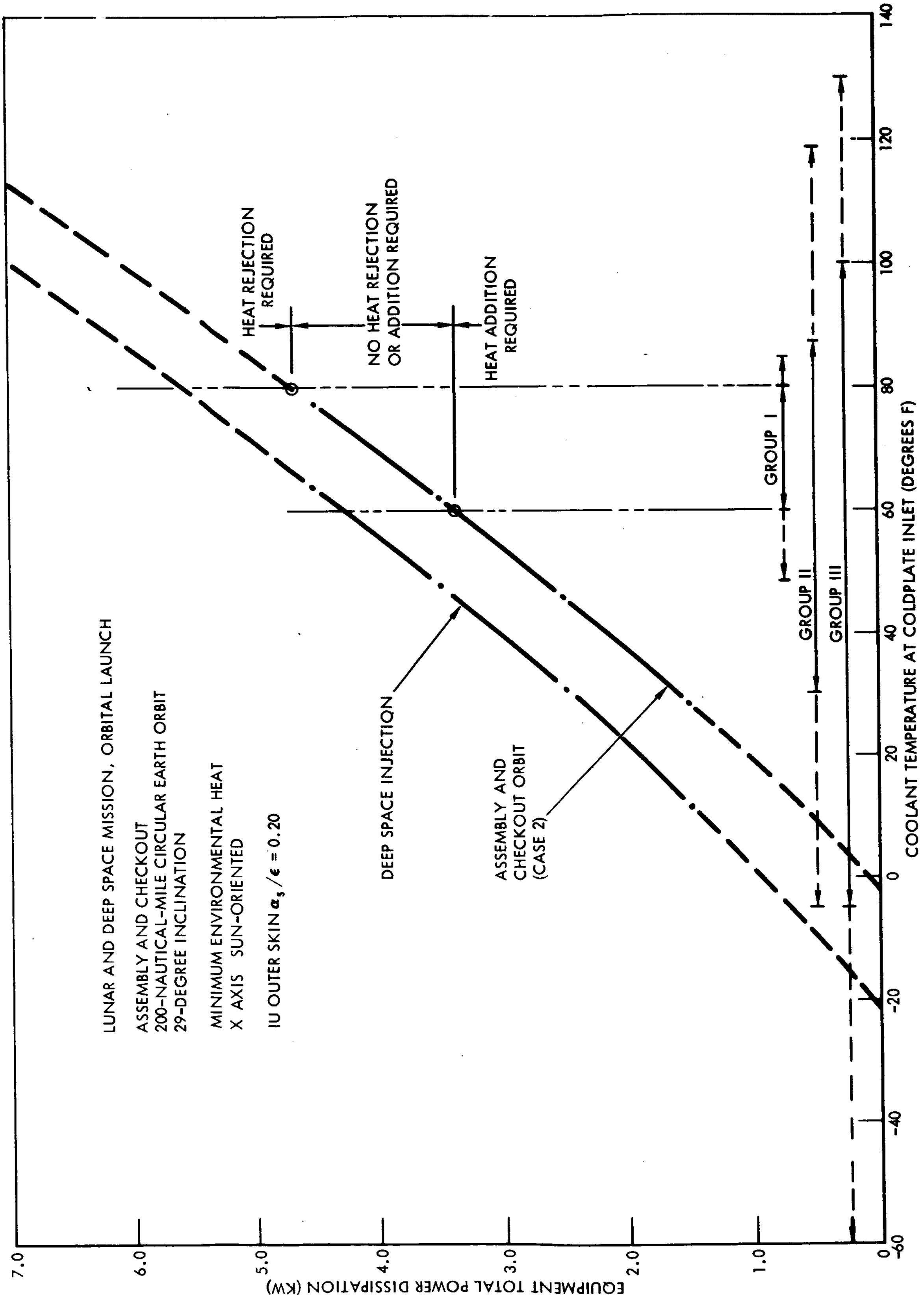


Figure 3-10. Coolant Temperature Versus Equipment Acceptable Total Power Dissipation

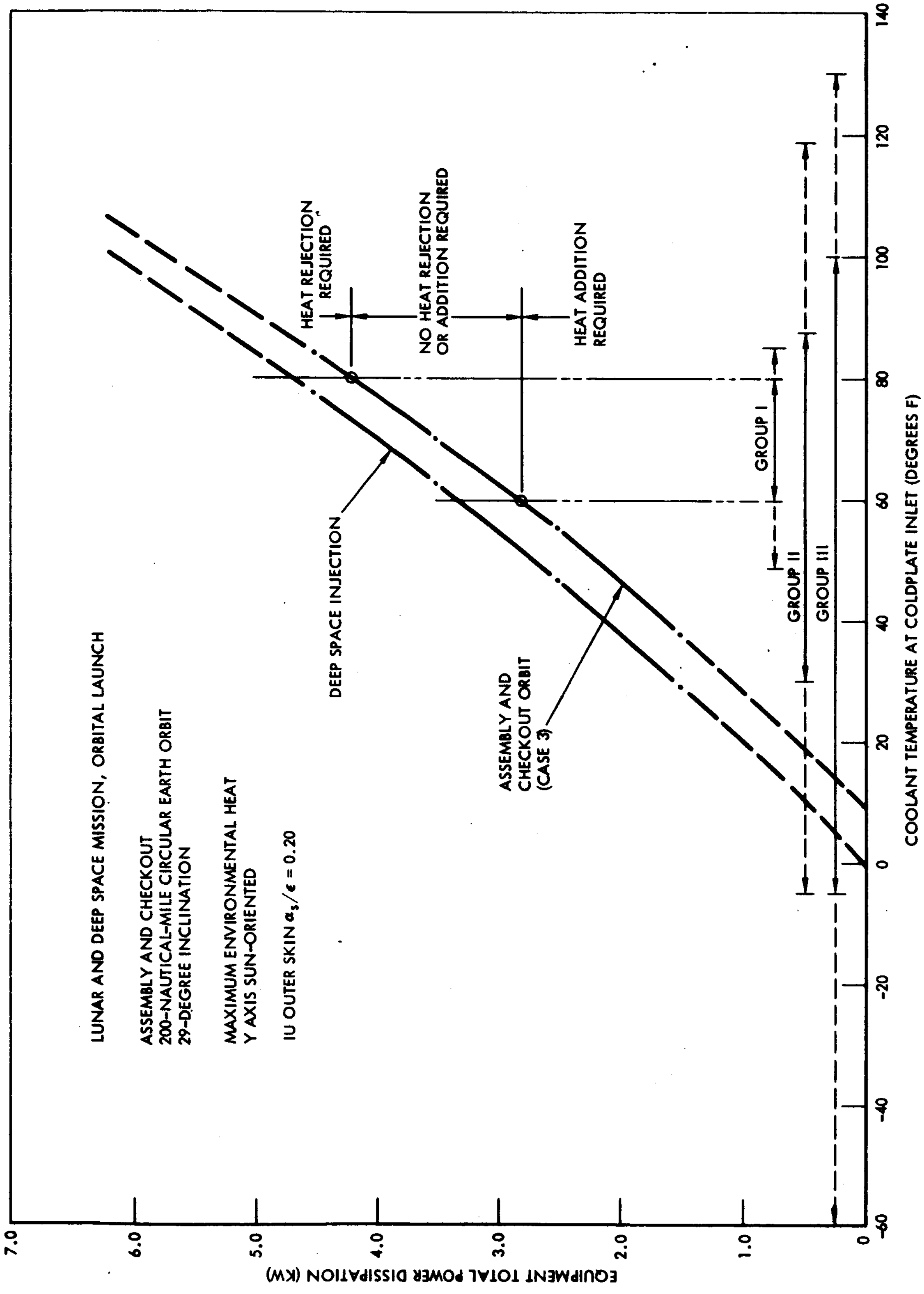


Figure 3-11. Coolant Temperature Versus Equipment Acceptable Total Power Dissipation

Table 3-4. Correlation Between Missions and Case Numbers

Fig. No.	Mission		Case No.	Appendix Fig. No.
3-5	200-n. mi. circular orbit 29° inclination	X axis sun-oriented	2	A-8
		Y axis sun-oriented	3	A-9
3-6	200 n. mi. polar orbit 90° inclination	X axis sun-oriented	5	A-8
		Y axis sun-oriented	6	A-10
3-7	Synchronous earth orbit (19,327)	X axis sun-oriented	9	A-11
		Y axis sun-oriented	8	A-8
3-8	Lunar, earth launch	Checkout orbit, 100-n. mi., earth orbit, 29° inclination	2	A-8
	Minimum environmental heat, x axis sun-oriented	Injection period	9	A-11
3-9	Lunar, earth launch	Checkout orbit, 100-n. mi. earth orbit, 29° inclination	3	A-9
	Maximum environmental heat, y axis sun-oriented	Injection period	8	A-8
3-10	Lunar and deep space, orbital launch Minimum environmental heat, x axis sun-oriented	Assembly and checkout orbit 200-n. mi. earth orbit, 29° inclination	2	A-8
		Injection period	9	A-11
3-11	Lunar and deep space, orbital launch Maximum environmental heat, y axis sun-oriented	Assembly and checkout orbit, 200-n. mi. earth orbit, 29° inclination	3	A-9
		Injection period	8	A-8

Figures 3-5 through 3-11 indicate the upper and lower limits of equipment power dissipation for the various missions and orbital conditions under consideration, thus avoiding the requirement for heat rejection from, or addition to, the coolant circuit. The criterion that establishes these equipment power dissipation limits is the permissible coolant temperature range determined from Figures 3-1 through 3-3. For example, in Figure 3-5 a Group I coolant temperature range of 60 F to 80 F is indicated by a solid horizontal line. Corresponding to this temperature range, and for the Case 3 orientation, the equipment total power dissipation upper and lower limits are 4. 2 and 2. 8 kilowatts, respectively. Likewise, the limits for Group II equipment are 4. 7 and 1. 1 kilowatts, respectively. These limits signify that, as long as the equipment total power dissipation remains within the limits, the resulting coolant temperature at coldplate inlet will satisfy equipment temperature requirements without heat rejection from, or addition to, the environmental control loop. The specific items of equipment that determine the permissible coolant temperature limits for each group are listed in Table 3-5; the corresponding equipment power dissipation limits for all missions under consideration are summarized in Table 3-6. It is apparent that requirements for all three groups of equipment are satisfied if the permissible power dissipation limits for Group I equipment are maintained. The effect of extending the equipment temperature limitations on permissible power dissipation levels is indicated by the dashed-line extensions of the group limits in Figures 3-5 through 3-11. Of course, satisfactory operation under present conditions can be obtained through suitable heat rejection or addition methods under all conditions of equipment power dissipation.

From the results of the thermal analysis presented in the appendix (SID 67-373-3) to this report, electronic package and coldplate temperatures at IU locations 2, 9, 15, and 20 and corresponding to the foregoing operating conditions have been summarized in Table 3-7. In this tabulation, electronic package temperatures falling within the Group I range of Figure 3-1 (50 F to 122 F) are indicated with asterisks. Examination of Table 3-7 reveals that the majority of the electronic package temperatures falls within this Group I range under the assumed operating conditions, and, with the exception of Cases No. 6 and 8 at α_s = 0. 90 and coldplate heat load of 250 watts, the remaining electronic package temperatures fall within the Group II range (-4 F to 167 F), which indicates that a relatively wide choice exists in regard to placement of electronic packages on available coldplates and that placement of most packages in a specific IU location is not critical. It has already been indicated that data for Cases 1, 4, 7, and 10 at coldplate heat loads of 50 and 250 watts were not obtained.

Table 3-5. Coolant Temperature Limits and Critical Electronic Packages

Group	Coolant Temperature Limits			
	NASA Specification		NAA Modification	
	Upper Limit (Figure 3-1)	Lower Limit (Figure 3-3)	Upper Limit (Figure 3-2)	Lower Limit (Figure 3-3)
I	80 F *ST-124M Inertial Platform (70 watts)	59 F Accelerometer Signal Conditioner (6.5 watts)	85 F *LVDA (400 watts)	49 F Accelerometer Signal Conditioner (6.5 watts)
II	87 F Azusa Transponder and Filter (270 watts)	30 F Telemeter Assembly, S1 (35 watts)	119 F RF Assembly-P1 (177 watts)	-5 F Telemeter Assembly, S1 (35 watts)
III	100 F Minstram Transponder (140 watts)	-5 F Remote Digital Multiplexer (4 watts)	130 F Minstram Transponder (140 watts)	-60 F Remote Digital Multiplexer (4 watts)

*Internal or integrally thermal conditioned equipment

Table 3-6. Equipment Power Dissipation Limits for Zero Net Heat Gain or Loss

Mission		Limit	Equipment Power Dissipation Limit (kw): Based on NASA Specification Temperature Limits			Based on NAA Modified Temperature Limits		
			Group I	Group II	Group III	Group I	Group II	Group III
200-n. mi. circular earth orbit 29° inclination	X axis sun-oriented (Case 2)	Upper	4.7	5.2	6.2	5.1	>7.0	>7.0
		Lower	3.4	1.6	0	2.7	0	0
	Y axis sun-oriented (Case 3)	Upper	4.2	4.7	5.7	4.6	>7.0	>7.0
		Lower	2.8	1.1	0	2.1	0	0
200-n. mi. polar earth orbit 90° inclination	X axis sun-oriented (Case 5)	Upper	4.7	5.2	6.1	5.0	>7.0	>7.0
		Lower	3.4	1.6	0	2.7	0	0
	Y axis sun-oriented (Case 6)	Upper	3.9	4.4	5.3	4.2	6.9	7.0
		Lower	2.5	0.8	0	1.9	0	0
Synchronous earth orbit (19,327 n. mi.)	X axis sun-oriented (Case 9)	Upper	5.6	6.1	6.9	5.9	>7.0	>7.0
		Lower	4.2	2.4	0.7	3.5	0.7	0
	Y axis sun-oriented (Case 8)	Upper	4.7	5.2	6.1	5.0	>7.0	>7.0
		Lower	3.3	1.6	0	2.6	0	0
Lunar, ground launch Minimum environmental heat, X axis sun-oriented	Checkout orbit 100-n. mi. earth orbit	Upper	4.7	5.2	6.0	5.0	>7.0	>7.0
		Lower	3.4	1.6	0	2.7	0	0
	Injection period	Upper	5.6	6.0	6.9	5.9	>7.0	>7.0
		Lower	4.3	2.4	0.7	3.6	0	0
Lunar, ground launch Maximum environmental heat, Y axis sun-oriented	Checkout orbit 100-n. mi. earth orbit	Upper	4.2	4.7	5.7	4.6	7.0	>7.0
		Lower	2.8	1.1	0	2.1	0	0
	Injection period	Upper	4.7	5.2	6.1	5.0	>7.0	>7.0
		Lower	3.3	1.6	0	2.6	0	0
Lunar and deep space orbital launch Minimum environmental heat X axis sun-oriented	Assembly and checkout period 200-n. mi. earth orbit 29° inclination	Upper	4.7	5.2	6.0	5.0	>7.0	>7.0
		Lower	3.4	1.6	0	2.7	0	0
	Injection period	Upper	5.6	6.1	6.9	5.9	>7.0	>7.0
		Lower	4.3	2.4	0.7	3.6	0	0
Lunar and deep space orbital launch Maximum environmental heat Y axis sun-oriented	Assembly and checkout period 200-n. mi. earth orbit 29° inclination	Upper	4.2	4.7	5.7	4.6	7.0	>7.0
		Lower	2.8	1.1	0	2.1	0	0
	Injection period	Upper	4.7	5.2	6.1	5.0	>7.0	>7.0
		Lower	3.3	1.6	0	2.6	0	0

The foregoing thermal analysis discussion is applicable to the time period when the S-IVB dome temperature has reached its equilibrium value of 60 F. To evaluate the effect of the S-IVB dome temperature on instrument unit net heat gain or loss during the initial phase of a mission, when this temperature is expected to be as low as -210 F, the effective emissivity of the dome was assumed to be controllable between values of 0. 05 and 0. 90. Instrument unit heat balances were obtained at the extremes of this emissivity range for the 200-nautical-mile earth orbit and with the y axis sun-oriented (Case 3). The electrical heat load was assumed to be 50 watts per coldplate, and coolant temperature at coldplate inlet was maintained at 30, 50, and 75 F, respectively. Results of the thermal analysis, which may be found in Volume 2 of this report, indicate that an effective dome emissivity of 0. 9 produces a significant heat loss from the instrument unit when the surface of the dome is at cryogenic temperatures. However, this heat loss can be suppressed quite readily by thermal isolation of the S-IVB dome through application of a cover having a low effective emissivity. This was confirmed by the results applicable to a dome emissivity of 0. 05, which reduces the heat loss to a more manageable level. The data suggest the possibility of using the S-IVB dome as a heat sink during the initial phase of a mission through selection of a dome cover with an appropriate infrared emissivity.

The effect of nonuniform coldplate heat dissipation on coolant heat gain or loss was investigated by analyzing the variation in heat flow to the coolant at each individual coldplate location. This analysis used as a starting point the coolant net heat gain or loss data of Figures A-4, A-5, and A-6 (in the appendix) for an IU outer skin solar absorptivity value of 0. 18. For equipment power dissipation rates of 50, 150, and 250 watts per coldplate, coolant heat gain or loss at each IU location was obtained from computer data printouts. These data are shown graphically in Figures 4-66 through 4-95 in Volume 2 of this report. Examination of the data suggests the possibility of obtaining the same result (zero net heat gain of the coolant) with different total equipment power dissipation rates depending on distribution of the power dissipation among the available coldplates.

To augment the above heat balances that are based on assumed values for equipment heat loads, additional heat balances were obtained which were based on consideration of the coldplate and electronics components mounted thereon as an adiabatic surface with no load on the electronic components. Heat balances were obtained for the current, or baseline, active thermal control system consisting of a closed liquid coolant loop with coldplates, electronic packages mounted on coldplates, and integrally or internally cooled electronic packages.

For the adiabatic surface case, the temperature of the coldplates and the electronic packages was assumed to be constant. Also, it was assumed that the coldplates and electronic packages were at the same temperature.

Case No.	Orbit Characteristics	IU Outer Skin, α	IU Location 2					
			Heat Load/Coldplate					
			50 Watts			150 Watts		
			30	50	75	30	50	75
IU-1a 1	200-n.mi. circ orbit Angle of incl: 29 deg Launch date: June 21 X axis along orbit path IU location 3 toward earth	0.18	←	NA	→	75* / 43	86* / 57	99* / 73
		0.90	←	NA	→	83* / 55	92* / 66	104* / 89
IU-1b 2	200-n.mi. circ orbit Angle of incl: 29 deg Launch date: June 21 X axis toward sun IU earth orientation varies with orbit position	0.18	36 / 21	47 / 35	62* / 52	66* / 33	78* / 47	91* / 63
		0.90	39 / 26	50* / 39	65* / 56	69* / 38	80* / 51	94* / 66
IU-1c 3	200-n.mi. circ orbit Angle of incl: 29 deg Launch date: June 21 Y axis toward sun IU location 21 toward sun	0.18	38 / 24	50* / 32	64* / 54	69* / 36	80* / 49	93* / 66
		0.90	49 / 36	61* / 50	75* / 67	79* / 48	90* / 61	104* / 78
IU-3a 4	Synchronous orbit Angle of incl: 0 deg Launch.date: March 21 X axis along orbit path IU location 3 toward earth	0.18	←	NA	→	78* / 47	88* / 59	100* / 75
		0.90	←	NA	→	92* / 65	103* / 78	107* / 92
IU-3b 5	200-n.mi. circ orbit Angle of incl: 29 deg Launch date: June 21 X axis toward sun IU earth orientation varies with orbit position	0.18	36 / 22	47 / 35	62* / 52	67* / 33	78* / 47	91* / 63
		0.90	36 / 22	48 / 35	62* / 52	67* / 33	78* / 47	91* / 63
IU-3c 6	200-n.mi. circ polar orbit Angle of incl: 90 deg Launch date: March 21 Y axis toward sun IU location 21 toward sun	0.18	40 / 27	52* / 41	66* / 57	71* / 39	82* / 52	95* / 69
		0.90	57* / 47	68* / 59	81* / 76	87* / 58	NA	NA
IU-4a 7	200-n.mi. circ polar orbit Angle of incl: 90 deg Launch date: March 21 X axis along orbit path IU location 3 toward earth	0.18	←	NA	→	76* / 45	87* / 58	99* / 73
		0.90	←	NA	→	83* / 53	92* / 67	NA
IU-6a 8	Synchronous orbit Angle of incl: 0 deg Launch date: March 21 Y axis toward sun IU location 21 toward sun	0.18	34 / 20	47 / 33	60* / 51	65* / 32	77* / 46	90* / 62
		0.90	50* / 40	61* / 53	75* / 70	82* / 52	92* / 66	NA
IU-6b 9	200-n.mi. circ polar orbit Angle of incl: 90 deg Launch date: March 21 X axis toward sun IU earth orientation varies with orbit position	0.18	30 / 14	42 / 29	57* / 46	61* / 28	72* / 41	87* / 57
		0.90	30 / 14	42 / 29	57* / 46	61* / 28	72* / 41	87* / 57
IU-6c 10	100-n.mi. circ equatorial orbit Angle of incl: 29 deg Launch date: March 21 X axis along orbit path IU location 3 toward earth	0.18	←	NA	→	73* / 40	83* / 52	97* / 69
		0.90	←	NA	→	109* / 79	118* / 90	NA

*Indicates electric package temperatures that fall within the Group I range of Figure 4-2 (50 F to 122 F)

NA = Not available

			IU Location 9												
			Heat Load/Coldplate												
250 Watts			50 Watts			150 Watts			250 Watts			50 Watts			
30	50	75	30	50	75	30	50	75	30	50	75	30	50	75	3
	NA			NA		67*/33	78*/47	92*/64		NA			NA		66*
	NA			NA		78*/48	87*/60	100*/82		NA			NA		83*
94*/44	105*/57	117*/73	37/22	48/36	62*/53	67*/34	78*/47	92*/64	95*/45	106*/58	118*/74	35/21	47/34	62*/52	66*
97*/48	108*/52	120*/77	40/27	52*/40	65*/56	70*/38	81*/51	95*/67	93*/49	108*/62	121/78	38/24	50*/38	64*/55	68*
96*/47	107*/59	119*/76	37/24	49/37	63*/54	68*/36	79*/49	92*/65	96*/47	106*/60	118*/76	38/23	50*/37	63*/54	69*
107*/61	117*/72	129/86	43/30	55*/48	69*/64	74*/46	85*/59	98*/74	102*/57	112*/69	124/84	50*/37	62*/52	75*/67	80*
	NA			NA		64*/30	76*/44	89*/60		NA			NA		67*
	NA			NA		67*/30	77*/44	89*/60		NA			NA		83*
94*/43	105*/57	118*/73	35/22	47/35	62*/52	66*/33	78*/47	92*/63	93*/43	105*/57	117*/73	35/22	47/35	62*/52*	66*
95*/44	105*/57	118*/73	36/22	48/35	62*/52	67*/33	78*/47	91*/63	94*/43	105*/57	117*/73	35/22	47/35	62*/52	66*
98*/48	109*/62	121*/77	35/20	47/34	61*/51	66*/32	77*/46	90*/62	93*/43	104*/56	117*/72	40/27	52*/41	66*/57	71*
113*/68	123/81	135/96	35/20	47/34	61*/51	66*/32	NA	NA	93*/43	104*/56	117*/72	58*/47	68*/60	82*/77	87*
	NA			NA		67*/34	78*/48	91*/63		NA			NA		68*
	NA			NA		74*/45	84*/57	NA		NA			NA		87*
93*/43	104*/57	117*/73	30/15	42/29	57*/46	61*/27	72*/40	87*/58	91*/38	100*/52	113*/68	35/20	47/34	60*/51	65*
109*/63	119*/75	132/91	30/15	42/29	57*/46	61*/28	72*/40	NA	89*/38	100*/51	113*/68	51*/41	62*/53	78*/71	82*
90*/39	100*/51	107*/68	30/14	42/29	57*/46	61*/28	72*/40	87*/58	90*/39	100*/51	107*/68	30/14	42/29	57*/46	61*
90*/39	100*/51	107*/68	30/14	42/29	57*/46	61*/28	72*/40	87*/57		NA			NA		61*
	NA			NA		61*/28	73*/41	87*/58		NA			NA		62*
	NA			NA		62*/28	74*/42	NA	90*/39	100*/51	107/68	30/14	42/29	57*/46	61*

Electronic Package Temperature → 38/24 ← Coldplate Temperature

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Table 3-7. Electronic Package and Coldplate Temperatures

IU Location 15 Heat Load/Coldplate						IU Location 20 Heat Load/Coldplate								
150 Watts			250 Watts			50 Watts			150 Watts			250 Watts		
p	50	75	30	50	75	30	50	75	30	50	75	30	50	75
35	78*/48	91*/65	←	NA	→	←	NA	→	65*/32	76*/45	90*/62	←	NA	→
62	93*/71	105*/87	←	NA	→	←	NA	→	70*/39	80*/51	93*/67	←	NA	→
33	77*/46	91*/63	94*/44	105*/57	117*/73	35/21	47/35	62*/52	66*/33	78*/47	91*/63	94*/44	105*/57	117*/73
37	80*/49	93*/66	96*/47	107*/60	119*/76	39/27	50*/40	64*/56	70*/38	81*/52	94*/67	97*/48	107*/62	120*/77
36	80*/49	93*/66	96*/47	107*/59	119*/76	46/33	57*/46	71*/62	76*/44	86*/58	99*/74	103*/54	113*/67	125/23
49	90*/62	105*/79	108*/60	118*/72	130/87	77*/67	90*/81	102*/95	105*/77	114*/89	129/107	132/85	141/100	152/114
32	78*/47	91*/62	←	NA	→	←	NA	→	80*/50	81*/63	104*/79	←	NA	→
52	94*/67	101*/79	←	NA	→	←	NA	→	129/110	137/121	148/135	←	NA	→
33	78*/47	91*/63	94*/43	105*/57	118*/73	35/21	47/34	61*/52	66*/33	77*/46	90*/62	94*/43	104*/57	117*/73
33	78*/47	91*/63	94*/43	105*/57	118*/73	35/21	47/34	61*/52	66*/32	77*/46	90*/62	94*/43	104*/57	117*/73
39	82*/52	95*/69	98*/48	109*/62	121*/78	47/40	63*/54	77*/69	82*/52	92*/65	105*/81	108*/62	118*/74	131/90
58	NA	NA	114*/68	123/81	135/97	102*/101	101*/102	123/127	129/111	NA	NA	153/119	162/130	172/144
35	79*/48	91*/65	←	NA	→	←	NA	→	67*/33	78*/47	91*/62	←	NA	→
60	85*/62	NA	←	NA	→	←	NA	→	75*/52	95*/72	NA	←	NA	→
32	78*/47	90*/62	93*/43	104*/57	117*/73	47/35	58*/48	71*/64	78*/48	89*/60	101*/77	105*/57	115*/70	127/86
51	93*/66	NA	110*/63	120*/76	132/92	94*/96	110*/110	120*/123	128/108	136/119	NA	151/116	160/128	170/141
23	72*/40	87*/58	89*/38	100*/51	107*/68	29/13	41/28	56*/44	60*/27	72*/40	86*/57	88*/38	99*/50	107*/68
28	73*/41	87*/57	←	NA	→	←	NA	→	60*/27	72*/40	87*/57	←	NA	→
28	74*/41	NA	←	NA	→	←	NA	→	62*/28	73*/41	NA	←	NA	→
28	72*/40	87*/58	89*/38	100*/51	107*/68	29/13	41/28	56*/44	60*/27	71*/40	86*/56	88*/38	99*/50	107*/68

Temperature values of 30, 60, and 75 F were used for the coldplates and the electronic packages mounted thereon, and for the internally cooled electronic packages.

The properties of the various surfaces involved in the heat transfer and the conduction between the coldplates and the instrument unit structure were considered the same as in the previous analysis. However, different values for fA's were used for the internal radiation heat transfer. This is the result of adding the radiation heat transfer between the IU structure at each coldplate location and the S-IVB dome and the SLA. This was not considered in the previous analysis because the amount of heat transferred is relatively small. The values for fA's are listed in Table 3-8, for two values for the emissivity of the S-IVB dome. For the coldplate and electronic packages mounted thereon, a single value for the fA's is given which was computed for a coldplate emissivity, of 0.18 and electronic packages emissivity of 0.90 (Table 3-2) and on the assumption that half of the coldplate surface was covered by the electronic packages.

To perform the thermal analysis with a digital computer, the thermal network used previously was modified to include the internal radiation heat transfer between the IU skin at each coldplate location and the S-IVB dome and between an assumed forward structure (SLA), and by eliminating the coolant flow network and the coldplate-mounted electronic package network. The coolant flow network was eliminated, since the coldplate temperature was maintained at a selected value, and thus coolant flow need not be included. The network for the coldplate-mounted electronic package was eliminated, since it was assumed that the coldplate and electronic packages mounted thereon were at the same temperature and thus both could be represented by a single node. The overall network for the instrument unit was modified to accommodate the changes. The assumptions made previously with regard to the construction and use of the thermal network were considered applicable.

Six orbital conditions were used to provide different environmental heat loads at various orbital altitudes and orbit inclinations. These conditions were considered to be the practical range of interest. Also, the vehicle orientation of X axis tangent to the flight path was considered to be the most significant.

The near-earth polar orbit (IU-3a) gives the maximum environmental heat load, since one side of the vehicle is continuously exposed to the sun, and the synchronous earth orbit (IU-6c) gives the minimum environmental heat load, since the earth emission and reflected solar energy are negligible and there is a considerable period of partial solar energy incident on the vehicle surface. The orbit identified as IU-7a is identical to the maximum shadow orbit in Reference 3-1, and orbit IU-10 is identical to the minimum

Table 3-8. Values for fA's (sq in.)

Component	Component/S-IVB Dome		Component/SLA	
	S-IVB Dome $\epsilon = 0.05$	S-IVB Dome $\epsilon = 0.90$	S-IVB Dome $\epsilon = 0.05$	S-IVB Dome $\epsilon = 0.90$
Coldplate/electronic package combined	34	318	126	48
ST-124 inertial platform	94	897	357	136
LVDA	74	708	280	107
LVDC	74	708	280	107
Flight control computer	74	708	280	107
IU skin (at coldplate location)	4	38	15	6

shadow orbit of Reference 3-1. Orbital cases IU-1a and IU-7a are identical except for the difference in the orbital altitude. The same is true for orbital cases IU-10 and IU-11.

Table 3-9 provides a convenient guide for the numerous heat balances for adiabatic coldplate surface conditions. This table gives the orbital data, S-IVB dome conditions, and the optical properties of the IU outer surfaces that were used. It is to be noted that for each of the 48 possible combinations indicated on the table, 3 coldplates/electronic package temperatures of 30, 60, and 75 F were used.

The results of the heat balances for the adiabatic coldplate surface condition are summarized in Tables 3-10, 3-11, and 3-12. These tables present the external heat gain or loss and internal heat gain or loss of the IU skin or coldplate/ electronic component for the configuration given in Figure 2-1, and total heat gain or loss as a result of heat transfer with space, the S-IVB dome, and the forward structure (SLA). A net heat gain indicates the amount of heat that must be absorbed by a heat rejection system, while a net heat loss indicates the amount of heat that must be added to the system to maintain the assumed coldplate/electronic equipment temperature. Through the use of these tables, the desired active thermal system parameters can be established based on the type of mission, equipment heat load profile and temperature tolerance, and S-IVB dome temperature profile. In each of the tables, the environmental heat absorbed and the IU skin temperature are averaged values for one orbit.

A review of the data presented in these tables indicates some general trends to be noted. First, the total heat gain or loss appears to vary linearly with the coldplate/electronic component temperature so that interpolations can be made with reasonable accuracy. Further, the data could be extrapolated to give some indication of the temperature value for the coldplate/electronic component for zero heat balance for zero equipment power dissipation. Second, minimum environmental heat absorption—i.e., low value for the absorptivity of IU outer surface—and minimum heat transfer between thermal conditioning system and the S-IVB dome appear desirable. These conditions are particularly desirable for the case of equipment heat rejection rate between two and three kilowatts, since this would minimize the need for either heat addition or heat rejection to maintain the desired equipment temperature. For the cases where the equipment heat dissipation rate is less than one kilowatt, high value for the absorptivity of the IU outer surface may be desirable in order to maintain the desired equipment temperature. Conversely, for the cases of high heat dissipation rate, four to five kilowatts, minimum environmental heat absorption and maximum heat transfer between thermal conditioning system and the S-IVB dome appear desirable, particularly if the S-IVB dome temperature is in the cryogenic temperature range (-210 F). Third,

Table 3-9. Orbital Data and Conditions for Heat Balances - Active System, Adiabatic Coldplate Surface

Case No. (Table 2-3)	Type of Orbit	Orientation	Orbital Altitude (n. mi.)	Launch Date	Angle of Inclination (deg)	S-IVB Dome Emissivity (ϵ)	S-IVB Dome Temp (F)	Outer Surface Optical Properties α_s	Outer Surface Optical Properties α_s/ϵ	Vol.2 Table No.
IU-1a	Circular earth orbit	X axis tangent to flight path	200	June 21	29	0.05	60	0.18 0.90	0.20 1.00	4-14
							-210	0.18 0.90	0.20 1.00	4-15
						0.90	60	0.18 0.90	0.20 1.00	4-16
							-210	0.18 0.90	0.20 1.00	4-17
IU-3a	Circular earth orbit (polar orbit)	X axis tangent to flight path	200	March 21	90	0.05	60	0.18 0.90	0.20 1.00	4-18
							-210	0.18 0.90	0.20 1.00	4-19
						0.90	60	0.18 0.90	0.20 1.00	4-20
							-210	0.18 0.90	0.20 1.00	4-21
IU-6c	Circular earth orbit (synchronous altitude)	X axis tangent to flight path	19,327	March 21	0	0.05	60	0.18 0.90	0.20 1.00	4-22
							-210	0.18 0.90	0.20 1.00	4-23
						0.90	60	0.18 0.90	0.20 1.00	4-24
							-210	0.18 0.90	0.20 1.00	4-25
IU-7a	Circular earth orbit	X axis tangent to flight path	100	June 21	29	0.05	60	0.18 0.90	0.20 1.00	4-26
							-210	0.18 0.90	0.20 1.00	4-27
						0.90	60	0.18 0.90	0.20 1.00	4-28
							-210	0.18 0.90	0.20 1.00	4-29
IU-10	Circular earth orbit (right ascension - 180 deg)	X axis tangent to flight path	100	Dec. 21	34	0.05	60	0.18 0.90	0.20 1.00	4-30
							-210	0.18 0.90	0.20 1.00	4-31
						0.90	60	0.18 0.90	0.20 1.00	4-32
							-210	0.18 0.90	0.20 1.00	4-33
IU-11	Circular earth orbit (right ascension - 180 deg)	X axis tangent to flight path	200	Dec. 21	34	0.05	60	0.18 0.90	0.20 1.00	4-34
							-210	0.18 0.90	0.20 1.00	4-35
						0.90	60	0.18 0.90	0.20 1.00	4-36
							-210	0.18 0.90	0.20 1.00	4-37

for the coldplate/electronic equipment temperature of 60 F, the total heat loss is nearly the same for the combination of low environmental heat absorption (IU outer surface absorptivity of 0.18) and low heat transfer with the S-IVB dome and for the combination of high environmental heat absorption and high heat transfer with the S-IVB dome.

Although the indicated trends are generally applicable for all orbital conditions considered, the determination of surface properties that greatly influence the heat transfer must still be made based on specific considerations. These are considered in the following section.

Table 3-10. Summary of Heat Balances—Active System, Adiabatic Coldplate Surface

Case No. (Table 2-3)	Type of Orbit and Orientation	Orbital Altitude (n. mi.)	Launch Date	Outer Skin Optical Properties α_s	Outer Skin Optical Properties α_s/ϵ	S-IVB Dome Emissivity (ϵ)	S-IVB Dome Temp (F)	Adiabatic Coldplate/Elec Component Temp (F)	External Heat Gain or Loss (-) of IU Skin (Btu/hr): Average Environ Heat Absorbed	External: IU Skin ↕ Space	External: Net	Internal Heat Gain or Loss (-) of IU Skin or Coldplate/Elec Comp (Btu/hr): IU Skin ↕ S-IVB Dome	Internal: IU Skin ↕ SLA	Internal: Coldplate/Elec Comp ↕ S-IVB Dome	Internal: Coldplate/Elec Comp ↕ SLA	Internal: Net	Total Heat Gain or Loss (-): Btu/hr	Total Heat Gain or Loss (-): kw
IU-1a	Circular earth orbit, angle of inclination 29 deg X axis tangent to flight path	200	June 21	0.18	0.20	0.05	60	30 60 75		-10981.2 -13075.7 -14204.5	-5748.9 -7843.4 -8972.2	19.4 12.3 8.5	42.6 16.0 1.7	158.9 0 -90.8	185.8 -405.3 -741.5	406.7 -377.0 -822.1	-5342.1 -8220.7 -9794.3	-1.57 -2.41 -2.87
							-210	30 60 75	6935.6	-10964.8 -13057.9 -14186.0	-5732.5 -7825.6 -9004.1	-33.1 -40.2 -44.0	42.8 16.2 1.9	-549.2 -708.1 -798.9	185.8 -405.3 -741.5	-353.9 -1137.4 -1582.5	-6086.4 -8963.0 -10536.2	-1.78 -2.63 -3.09
						0.90	60	30 60 75	33.3 Btu/(hr)(sq ft)	-11019.4 -13105.6 -14229.2	-5787.1 -7873.3 -8996.9	182.8 115.4 79.2	16.8 6.1 0.4	1492.0 0 -849.4	71.2 -155.3 -283.1	1762.8 -33.8 -1052.9	-4024.1 -7907.1 -10049.6	-1.18 -2.32 -2.95
							-210	30 60 75		-10869.1 -12941.9 -14059.0	-5636.8 -7709.6 -8826.7	-311.5 -378.5 -414.5	17.5 7.0 1.3	-5184.1 -6676.0 -7525.4	71.2 -155.3 -283.1	-5406.9 -7202.8 -8221.7	-11043.7 -14912.4 -17048.4	-3.24 -4.37 -5.00
				0.90	1.0	0.05	60	30 60 75		-14047.9 -16347.6 -17576.3	103.8 -2195.9 -3424.6	11.3 3.5 -0.6	12.3 -17.0 -32.7	158.9 0 -90.8	185.8 -405.3 -741.5	368.3 -418.8 -865.6	472.2 -2614.8 -4290.2	0.14 -0.77 -1.26
							-210	30 60 75	17005.7	-14030.3 -16328.6 -17556.6	121.4 -2176.9 -3404.9	-41.1 -49.0 -53.1	12.5 -16.8 -32.4	-549.2 -708.1 -798.9	185.8 -405.3 -741.5	-392.2 -1179.2 -1625.9	-270.8 -3356.1 -5030.8	-0.08 -0.98 -1.47
						0.90	60	30 60 75	81.7 Btu/(hr)(sq ft)	-14078.9 -16365.5 -17586.7	72.8 -2213.8 -3435.0	106.7 32.6 -6.7	4.8 -6.9 -13.2	1492.0 0 -849.4	71.2 -155.3 -283.1	1674.7 -129.6 -1152.6	1747.6 -2343.0 -4587.6	0.51 -0.69 -1.34
							-210	30 60 75		-13917.0 -16191.0 -17406.0	234.7 -2039.3 -3254.3	-387.1 -460.8 -500.1	5.6 -6.0 -12.3	-5184.1 -6676.0 -7525.4	71.2 -155.3 -283.1	-5494.4 -7289.1 -8320.9	-5259.7 -9337.4 -11575.2	-1.54 -2.74 -3.39
IU-3a	Circular earth orbit (polar), angle of inclination 90 deg X axis tangent to flight path	200	Mar 21	0.18	0.20	0.05	60	30 60 75		-12255.2 -14384.1 -15529.6	-4659.3 -6788.2 -7933.7	18.1 10.9 7.0	37.6 10.5 -4.0	158.9 0 -90.8	185.8 -405.3 -741.5	400.4 -383.9 -829.3	-4258.9 -7172.1 -8763.0	-1.25 -2.10 -2.57
							-210	30 60 75	9671.8	-12239.0 -14366.4 -15511.2	-4643.1 -6770.5 -7915.3	-34.4 -41.6 -45.5	37.8 10.8 -3.7	-549.2 -708.1 -798.9	185.8 -405.3 -741.5	-360.0 -1144.2 -1589.6	-5003.1 -7914.7 -9504.9	-1.47 -2.32 -2.79
						0.90	60	30 60 75	46.5 Btu/(hr)(sq ft)	-12293.4 -14412.8 -15552.5	-4697.5 -6816.9 -7956.6	170.3 101.9 65.2	14.8 4.0 -1.8	1492.0 0 -849.4	71.2 -155.3 -283.1	1748.3 -49.4 -1069.1	-2949.2 -6866.3 -9025.7	-0.86 -2.01 -2.65
							-210	30 60 75		-12144.5 -14250.6 -15383.9	-4548.6 -6654.7 -7788.0	-324.0 -391.9 -428.4	15.6 4.8 -0.9	-5184.1 -6676.0 -7525.4	71.2 -155.3 -283.1	-5421.3 -7218.4 -8237.8	-9969.9 -13873.1 -16025.8	-2.92 -4.07 -4.70
				0.90	1.0	0.05	60	30 60 75		-21163.1 -23582.7 -24868.8	4803.8 2384.2 1098.1	2.5 -5.7 -10.1	-20.7 -51.6 -68.0	158.9 0 -90.8	185.8 -405.3 -741.5	326.5 -462.6 -910.4	5130.3 1921.6 187.7	1.50 0.56 0.06
							-210	30 60 75	30684.5	-21144.9 -23563.2 -24848.6	4822.0 2403.7 1118.3	-49.9 -58.2 -62.5	-20.5 -51.3 -67.7	-549.2 -708.1 -798.9	185.8 -405.3 -741.5	-433.8 -1222.9 -1670.6	4388.2 1180.8 -552.3	1.29 0.35 -0.16
						0.90	60	30 60 75	147.5 Btu/(hr)(sq ft)	-21175.7 -23577.3 -24853.5	4791.2 2389.6 1113.4	25.1 -52.8 -94.1	-8.1 -20.4 -27.0	1492.0 0 -849.4	71.2 -155.3 -283.1	1580.2 -228.5 -1253.6	6371.4 2161.1 -140.2	1.87 0.63 -0.04
							-210	30 60 75		-21009.6 -23399.0 -24669.4	4957.3 2567.9 1297.5	-468.5 -545.9 -587.0	-7.3 -19.5 -26.0	-5184.1 -6676.1 -7525.4	71.2 -155.3 -283.1	-5588.7 -7396.7 -8421.5	-631.4 -4828.8 -7124.0	-0.19 -1.42 -2.09

Table 3-11. Summary of Heat Balances—Active System, Adiabatic Coldplate Surface

Case No. (Table 2-3)	Type of Orbit and Orientation	Orbital Altitude (n. mi.)	Launch Date	Outer Skin Optical Properties: α_s	Outer Skin Optical Properties: α_s/ϵ	S-IVB Dome: Emissivity (ϵ)	S-IVB Dome: Temp (F)	Adiabatic Coldplate/Elec Component Temp (F)	External Heat Gain or Loss (-) of IU Skin (Btu/hr): Average Environ Heat Absorbed	External Heat Gain or Loss (-) of IU Skin (Btu/hr): IU Skin ↕ Space	External Heat Gain or Loss (-) of IU Skin (Btu/hr): Net	Internal Heat Gain or Loss (-) of IU Skin or Coldplate/Elect Comp (Btu/hr): IU Skin ↕ S-IVB Dome	Internal: IU Skin ↕ SLA	Internal: Coldplate/Elec Comp ↕ S-IVB Dome	Internal: Coldplate/Elec Comp ↕ SLA	Internal: Net	Total Heat Gain or Loss (-): Btu/hr	Total Heat Gain or Loss (-): kw
IU-6c	Circular earth orbit (synchronous), angle of inclination 0 deg	19,327	Mar 21	0.18	0.20	0.05	60	30	3085.1	-10135.4	-7496.7	21.6	51.4	158.9	185.8	417.9	-7078.8	-2.07
	X axis tangent to flight path							60	14.8 Btu/(hr)(sq ft)	-12165.2	-9526.5	14.9	25.6	0.0	-405.3	-364.8	-9891.3	-2.90
								75		-13262.6	-10623.9	11.2	11.7	-90.8	-741.5	-809.4	-11433.3	-3.35
							-210	30		-10118.8	-7479.9	-30.7	51.6	-549.2	185.8	-342.5	-7821.4	-2.29
								60		-12147.1	-9508.2	-37.6	25.8	-708.1	-405.3	-1125.2	-10633.4	-3.12
								75		-13243.7	-10604.8	-41.3	11.9	-798.9	-741.5	-1569.8	-12174.6	-3.57
						0.90	60	30		-10183.6	-7544.7	204.7	20.2	1492.0	71.2	1788.1	-5756.6	-1.69
								60		-12206.4	-9567.5	139.4	9.9	0.0	-155.3	6.0	-9573.5	-2.81
								75		-13299.1	-10660.2	104.2	4.4	-849.4	-283.1	-1023.9	-11634.1	-3.42
							-210	30		-10030.7	-7391.8	-289.6	21.0	-5184.1	71.2	-5381.5	-12773.3	-3.74
								60		-12039.7	-9400.8	-354.4	10.8	-6676.0	-155.3	-7174.9	-16575.7	-4.86
								75		-13125.8	-10486.9	-389.4	5.2	-7525.4	-283.1	-8192.7	-18679.6	-5.47
				0.90	1.0	0.05	60	30	15425.5	-13715.6	-523.2	12.2	15.7	158.9	-185.8	372.6	-150.6	-0.04
								60	74.2 Btu/(hr)(sq ft)	-15992.2	-2799.8	4.5	-13.4	0.0	-405.3	-414.2	-3214.0	-0.94
								75		-17209.7	-4017.3	0.4	-28.9	-90.8	-741.5	-860.8	-4878.1	-1.43
							-210	30		-13697.9	-505.5	-40.2	15.9	-549.2	185.8	-387.7	-893.2	-0.26
								60		-15973.1	-2780.7	-48.0	-13.1	-708.1	-405.3	-1174.5	-3955.2	-1.16
								75		-17189.9	-3997.5	-52.1	-28.6	-798.9	-741.5	-1459.7	-5618.6	-1.65
						0.90	60	30		-13749.9	-557.5	115.1	6.1	1492.0	71.2	1684.4	1126.9	0.33
								60		-16014.0	-2821.6	41.7	-5.5	0.0	-155.3	-119.1	-2940.7	-0.86
								75		-17224.2	-4031.8	2.6	-11.9	-849.4	-283.1	-1141.8	-5173.6	-1.52
							-210	30		-13587.4	-395.0	-378.8	6.9	-5184.1	71.2	-5484.8	-5879.8	-1.72
								60		-15838.8	-2646.4	-451.7	-4.6	-6676.0	-155.3	-7287.6	-9934.0	-2.91
								75		-17042.8	-3850.4	-490.6	-10.8	-7525.4	-283.1	-8309.9	-12160.3	-3.56
IU-7a	Circular earth orbit, angle of inclination 29 deg	100	June 21	0.18	0.20	0.05	60	30	7464.4	-11114.7	-5509.6	19.1	41.4	158.9	185.8	405.2	-5104.0	-1.50
	X axis tangent to flight path							60	35.9 Btu/(hr)(sq ft)	-13217.9	-7612.8	12.0	14.7	0.0	-405.3	-378.6	-7991.5	-2.34
								75		-14350.9	-8745.8	8.1	0.3	-90.8	-741.5	-823.9	-9569.3	-2.80
							-210	30		-11098.4	-5493.3	-33.4	41.6	-549.2	185.8	-355.2	-5848.5	-1.71
								60		-13200.1	-7595.0	-40.5	14.9	-708.1	-405.3	-1139.0	-8734.0	-2.56
								75		-14332.4	-8727.3	-44.3	0.5	-798.9	-741.5	-1584.2	-10311.5	-3.02
						0.90	60	30		-11151.9	-5546.8	179.8	16.3	1492.0	71.2	1759.3	-3787.5	-1.11
								60		-13246.5	-7641.4	112.2	5.6	0.0	-155.3	-37.5	-7678.9	-2.25
								75		-14374.3	-8769.2	75.8	-0.1	-849.4	-283.1	-1056.8	-9826.0	-2.88
							-210	30		-11001.6	-5396.5	-314.5	17.1	-5184.1	71.2	-5410.3	-10806.9	-3.17
								60		-13082.9	-7477.8	-381.7	6.4	-6676.0	-155.3	-7206.6	-14684.4	-4.30
								75		-14204.0	-8598.9	-417.8	0.7	-7525.4	-283.1	-8225.6	-16824.4	-4.93
				0.90	1.0	0.05	60	30	17702.6	-14246.1	411.7	10.9	10.7	158.9	185.8	366.3	779.6	0.23
								60	85.1 Btu/(hr)(sq ft)	-16556.6	-2318.3	3.0	-18.8	0.0	-405.3	-421.1	-2739.4	-0.80
								75		-17790.6	-3132.8	-1.1	-34.5	-90.8	-741.5	-859.5	-4000.7	-1.17
							-210	30		-14228.4	429.4	-41.6	10.9	-549.2	185.8	-394.1	35.1	0.01
								60		-16537.6	-1879.8	-49.4	-18.6	-708.1	-405.3	-1181.4	-3061.2	-0.90
								75		-17770.8	-3113.0	-53.6	-34.2	-798.9	-741.5	-1628.2	-4741.2	-1.39
						0.90	60	30		-14275.8	382.0	102.6	4.1	1492.0	71.2	1669.9	2051.9	0.60
								60		-16573.1	-1915.3	28.1	-7.7	0.0	-155.3	-134.9	-2050.2	-0.60
								75		-17799.3	-3141.5	-11.6	-13.9	-849.4	-283.1	-1158.0	-4299.5	-1.26
							-210	30		-14113.8	544.0	-391.3	4.9	-5184.1	71.2	-5499.3	-4955.3	-1.45
								60		-16398.4	-1740.6	-465.3	-6.8	-6676.0	-155.3	-7303.4	-9044.0	-2.65
								75		-17618.6	-2960.8	-504.8	-13.0	-7525.4	-283.1	-8326.3	-11287.1	-3.31

Table 3-12. Summary of Heat Balances—Active System, Adiabatic Coldplate Surface

Case No. (Table 2-3)	Type of Orbit and Orientation	Orbital Altitude (n. mi.)	Launch Date	Outer Skin Optical Properties		S-IVB Dome		Adiabatic Coldplate/Elec Component Temp (F)	External Heat Gain or Loss (-) of IU Skin (Btu/hr)			Internal Heat Gain or Loss (-) of IU Skin or Coldplate/Elec Comp (Btu/hr)					Total Heat Gain or Loss (-)	
				α_s	ϵ_s	Emissivity (ϵ)	Temp (F)		Average Environ Heat Absorbed	IU Skin → Space	Net	IU Skin → S-IVB Dome	IU Skin → SLA	Coldplate/Elec Comp → S-IVB Dome	Coldplate/Elec Comp → SLA	Net	Btu/hr	kw
IU-10	Circular earth orbit (right ascension 180 deg), angle of inclination 34 deg	100	Dec 21	0.18	0.20	0.05	60	30	8430.8	-11088.7	-5268.8	18.7	39.8	158.9	185.8	403.2	-4865.6	-1.43
	X axis tangent to flight path							60	40.5 Btu/(hr)(sq ft)	-13200.7	-7380.8	11.5	12.9	0	-405.3	-380.9	-7761.7	-2.27
								75		-14338.2	-8518.3	7.6	-1.6	-90.8	-741.5	-826.3	-9344.4	-2.74
							-210	30		-11072.2	-5252.3	-33.8	40.0	-549.2	185.8	-357.2	-5609.5	-1.64
								60		-13182.9	-7363.0	-41.0	13.1	-708.1	-405.3	-1141.3	-8504.3	-2.49
								75		-14319.6	-8499.7	-44.8	-1.3	-798.9	-741.5	-1586.5	-10086.2	-2.96
						0.90	60	30		-11121.7	-5301.8	175.8	15.7	1492.0	71.2	1754.7	-3546.9	-1.04
								60		-13225.1	-7405.2	107.7	4.9	0	-155.3	-42.7	-7448.1	-2.18
								75		-14357.1	-8537.2	71.2	-0.9	-849.0	-283.1	-1062.2	-9599.4	-2.81
							-210	30		-10971.2	-5151.3	-318.5	16.4	-5184.1	71.2	-5415.0	-10566.3	-3.10
								60		-13061.4	-7241.5	-386.1	5.8	-6676.0	-155.3	-7211.6	-14453.1	-4.24
								75		-14186.9	-8367.0	-422.4	0.0	-7525.4	-283.1	-8230.9	-16597.9	-4.86
				0.90	1.0	0.05	60	30	22535.8	-14291.4	1441.6	7.9	0.5	158.9	185.8	353.1	1794.7	0.53
								60	108.3 Btu/(hr)(sq ft)	-16641.4	-908.4	-0.1	-30.6	0	-405.3	-436.0	-1344.8	-0.39
								75		-17894.2	-2161.2	-4.4	-46.6	-90.8	-741.5	-883.3	-3044.8	-0.89
							-210	30		-14273.4	1459.6	-44.6	-0.3	-549.2	185.8	-408.3	1051.3	0.31
								60		-16622.0	-889.0	-52.6	-30.4	-708.1	-405.3	-1196.4	-2085.4	-0.61
								75		-17874.2	-2141.2	-56.8	-46.4	-798.9	-741.5	-1643.6	-3784.8	-1.11
						0.90	60	30		-14293.7	1439.3	74.8	-0.3	1492.0	71.2	1637.7	3077.2	0.90
								60		-16629.9	-896.9	-1.2	-12.3	0	-155.3	-168.8	-1065.7	-0.31
								75		-17874.8	-2141.8	-41.6	-18.7	-849.4	-283.1	-1192.8	-3334.5	-0.98
							-210	30		-14128.9	1604.1	-419.0	0.6	-5184.1	71.2	-5531.3	-3927.2	-1.15
								60		-16452.8	-719.8	-494.5	-11.4	-6676.0	-155.3	-7337.2	-8057.0	-2.36
								75		-17691.6	-1958.6	-534.7	-17.7	-7525.4	-283.1	-8360.9	-10319.6	-3.02
IU-11	Circular earth orbit (right ascension 180 deg), angle of inclination 34 deg	200	Dec 21	0.18	0.20	0.05	60	30	8144.8	-10995.3	-5411.7	18.9	40.5	158.9	185.8	404.1	-5007.7	-1.47
	X axis tangent to flight path							60	39.2 Btu/(hr)(sq ft)	-13102.3	-7518.7	11.7	13.6	0	-405.3	-380.0	-7898.8	-2.32
								75		-14237.3	-8653.7	7.9	-0.8	-90.8	-741.5	-825.2	-9478.9	-2.78
							-210	30		-10978.9	-5395.3	-33.6	40.7	-549.2	185.8	-356.3	-5751.6	-1.69
								60		-13084.5	-7500.9	-40.8	13.9	-708.1	-405.3	-1140.3	-8641.2	-2.53
								75		-14218.7	-8635.1	-44.6	-0.6	-798.9	-741.5	-1585.6	-10220.7	-3.00
						0.90	60	30		-11028.8	-5445.2	177.5	15.9	1492.0	71.2	1756.6	-3688.6	-1.08
								60		-13127.2	-7543.6	109.6	5.2	0	-155.3	-40.5	-7585.2	-2.22
								75		-14256.8	-8673.2	73.1	-0.6	-849.4	-283.1	-1060.0	-9733.2	-2.85
							-210	30		-10878.2	-5294.6	-316.8	16.7	-5184.1	71.2	-5413.0	-10707.4	-3.14
								60		-12963.3	-7379.7	-384.3	6.0	-6676.0	-155.3	-7209.6	-14588.6	-4.28
								75		-14086.4	-8502.8	-420.5	0.3	-7525.4	-283.1	-8228.7	-16731.6	-4.90
				0.90	1.0	0.05	60	30	23211.6	-14337.9	1729.5	7.5	-2.0	158.9	185.8	350.2	2079.7	0.61
								60	111.6 Btu/(hr)(sq ft)	-16692.3	-624.9	-0.5	-32.1	0	-405.3	-437.9	-1062.8	-0.31
								75		-17947.2	-1879.8	-4.8	-48.1	-90.8	-741.5	-885.2	-2765.0	-0.81
							-210	30		-14319.8	1747.6	-44.9	-1.7	-549.2	185.8	-410.0	1337.6	0.39
								60		-16672.9	-605.5	-53.0	-31.9	-708.1	-405.3	-1189.3	-1803.8	-0.53
								75		-17927.1	-1859.7	-57.2	-47.9	-798.9	-741.5	-1645.5	-3505.2	-1.03
						0.90	60	30		-14337.5	1729.9	71.2	0.8	1492.0	71.2	1635.2	3363.5	0.99
								60		-16678.0	-610.6	-4.9	-12.9	0	-155.3	-173.1	-783.4	-0.23
								75		-17924.9	-1857.5	-45.3	-19.3	-849.4	-283.1	-1197.1	-3054.6	-0.90
							-210	30		-14172.0	1895.4	-422.5	0.0	-5184.1	71.2	-5535.4	-3639.2	-1.07
								60		-16500.2	-432.8	-498.2	-12.0	-6676.0	-155.3	-7341.5	-7773.3	-2.28
								75		-17741.1	-1673.7	-538.5	-18.3	-7525.4	-283.1	-8365.3	-10039.0	-2.94

4.0 RECOMMENDED CONCEPTS

Preceding sections of this report have established the environmental control requirements for electronic packages on the IU under various orbital conditions. The selection of applicable TCS concepts to provide the required thermal control must consider not only the permissible equipment temperature range, but also the length of time during which the system is expected to function. To that end, several probable missions have been postulated and are listed in Table 4-1, together with an estimate of respective minimum and maximum durations. Based on these missions, coolant temperature limits established in the previous section, and equipment power dissipation-coolant temperature relationships (Figures 3-5 through 3-11), possible thermal conditioning system concepts are described in the following paragraphs, together with a discussion on the applicability to the postulated missions and durations.

Table 4-1. Estimate of Mission Duration

Mission	Duration (hours)	
	Minimum	Maximum
Earth Orbit	4-1/2	4320 (180 days)
Synchronous Orbit (19, 327 Nautical Miles)	12	4320 (180 days)
Lunar (Ground Launch)	4-1/2	24
Lunar and Deep Space (Orbital Launch)	720 (30 days)	1440 (60 days)

Examination of the data defining coolant temperature limits and data on the effect of variation in equipment power dissipation on coolant temperature reveals that only a few items of electronic equipment appear to exert an undue influence on thermal control requirements. These specific electronic packages are identified in Table 4-2, in accordance with the coolant temperature limits that they impose on each equipment group. For example, the lower temperature limit for Group I equipment is determined by the 59 F

Table 4-2. **Coolant Temperature Limits and Critical Electronic Packages**

Group	**Coolant Temperature Limits (degrees)**			
	NASA Specification		**NAA Modification**	
	Upper Limit (Figure 3-1)	**Lower Limit** (Figure 3-3)	**Upper Limit** (Figure 3-2)	**Lower Limit** (Figure 3-3)
I	**80 F** ST-124M Inertial **Platform*** (70 watts)	59 F Accelerometer Signal Conditioner (65 watts)	**85 F** **LVDA*** (400 watts)	**49 F** **Accelerometer Signal Conditioner** (65 watts)
II	87F Azusa Transponder and Filter (270 watts)	30 F **Telemeter** Assembly, S1 (35 watts)	**119 F** **RF Assembly-P1** (177 watts)	**-5 F** **Telemeter** (35 watts)
III	100 F Minstram **Transponder** (140 watts)	-5 F Remote Digital **Multiplexer** (4 watts)	130 F Minstram **Transponder** (140 watts)	-60 F Remote Digital Multiplexer (4 watts)

***Internal or integrally thermal-conditioned equipment**

requirement of the accelerometer signal conditioner, based on present NASA specifications. The upper temperature limit for Group I equipment is determined by the 80 F requirement of the ST-124M inertial platform assembly. Hence, if the thermal conditioning system were required to satisfy these temperature limits for Group I electronic packages without active removal or addition of heat, the equipment power dissipation of all equipment would have to be such that the 59 to 80 F coolant temperature range at coldplate inlet is achieved by purely passive means. Allowable equipment power dissipation rates are listed in Table 3-5 for the different mission parameters under consideration. This tabulation shows that, if coolant temperature requirements for equipment in Group I are met, Group II and Group III equipment requirements also are satisfied without any additional control system. Table 3-5 also indicates that the present instrument unit total power dissipation of 3.9 kilowatts falls within permissible Group I limits for all listed missions and vehicle orientations except the synchronous orbit and lunar and deep-space injection periods under minimum environmental heating conditions (X axis, sun-oriented).

If a separate provision were to be made for thermal control of the ST-124M inertial platform assembly, the 80 F upper coolant temperature limit for Group I equipment would be extended to the 85 F requirement of the launch vehicle data adapter (LVDA). Similarly, if the accelerometer signal conditioner, the platform AC power supply, and the ST-124 electronics were to be provided with a means of preventing them from being overcooled, the lower coolant temperature limit for Group I equipment could be decreased to 40 F. (This is based on the assumption that the separate thermal control for the ST-124M inertial platform assembly is capable of maintaining the temperature of this assembly at or above its present 50 F lower limit.) The resultant band of permissible coolant temperatures (40 F to 85 F) could be maintained over a wider range of equipment power dissipation and mission profiles without the requirement of active heat rejection or addition. The permissible power dissipation rates, as well as those resulting from a widening of the permissible coolant temperature range that is possible through equipment modifications as discussed in Section 3.0, are summarized in Table 4-3.

The addition of separate controls for critical items of equipment will increase the complexity of the TCS, which may not be justified for short-duration missions. Therefore, assuming that it is desirable to restrict the number of modifications to the present IU thermal conditioning system to a minimum, it is suggested that only a means for adding heat to the coolant circuit be incorporated into the present system for mission durations of less than 24 hours. This suggestion applies to near-earth orbital missions, synchronous orbit missions, and the earth-orbit phase of ground-launched lunar missions. The justification for this suggestion is that appreciable radiation heat transfer takes place from the IU to the S-IVB dome during the

Table 4-3. Equipment Power Dissipation Limits for Zero Net Heat Gain or Loss

Case No.	Mission	Equipment Power Dissipation Limits (kw)							
		Based on NASA Specification Temperature Limits				Based on NAA Modified Temperature Limits			
		Without Coolant Flow Restriction		With Coolant Flow Restriction		Without Coolant Flow Restriction		With Coolant Flow Restriction	
		Lower Limit	Upper Limit	Lower Limit	Upper Limit	Lower Limit	Upper Limit	Lower Limit	Upper Limit
IU-1b-2	200-n. mi. circular earth orbit 29-deg inclination X axis sun-oriented	3.4	4.7	2.1	4.7	2.7	5.1	1.3	5.1
IU-1c-3	200-n. mi. circular earth orbit 29-deg inclination Y axis sun-oriented	2.8	4.2	1.6	4.2	2.1	4.6	0.8	4.6
IU-3b-5	200-n. mi. polar earth orbit 90-deg inclination X axis sun-oriented	3.4	4.7	2.1	4.7	2.7	5.0	1.3	5.0
IU-3c-6	200-n. mi. polar earth orbit 90-deg inclination Y axis sun-oriented	2.5	3.9	1.3	3.9	1.9	4.2	0.5	4.2
IU-6a-8	Synchronous earth orbit 0-deg inclination Y axis sun-oriented	3.3	4.7	2.1	4.7	2.6	5.0	1.3	5.0
IU-6b-9	Synchronous earth orbit 0-deg inclination X axis sun-oriented	4.2	5.6	3.0	5.6	3.5	5.9	2.2	5.9

first few hours of orbit time, when the dome is still at cryogenic temperatures, unless the IU is thermally isolated from the dome. Such thermal isolation may be achieved by covering the S-IVB dome with a low-emissivity shield such as aluminized mylar. In the event that the IU is not thermally isolated from the S-IVB dome, a quantity of heat must be supplied at a rate decreasing with orbit time in order to maintain the nominal 60 F coolant temperature at coldplate inlet during the initial orbit period. The required amount of heating could be furnished by electrical resistance heating elements incorporated in the coolant inlet manifold, and, when heat removal from the system becomes necessary, the existing heat sink (sublimator) could be utilized. Of course, the weight of expendables (water) for a 24-hour mission will have to be evaluated against possible alternatives. It is also suggested that coolant flow can be restricted to a rate of 60 pounds per hour per coldplate. This rate was used for the major portion of the thermal analysis and was found to be satisfactory.

The separate thermal control for the ST-124M inertial platform assembly, mentioned above, can be achieved most conveniently by using a thermoelectric device that supplies heating as well as cooling. In the event that the coolant temperature at the inlet of the inertial platform assembly exceeds the maximum permissible for the assembly, the thermoelectric device serves to lower the temperature of the portion of the coolant that flows through the assembly. Implied in this situation is an arrangement, similar to that shown in Figure 4-1, in which the coolant passes over the cold surface of the thermoelectric device before it enters the assembly. The heat removed from the coolant by the thermoelectric device, plus the heat equivalent of the device power input, is rejected to the main coolant supply through suitable routing of some coolant flow past the hot surface of the device. When the coolant temperature at platform assembly inlet is below the minimum permissible for the assembly, the heat flow direction across the thermoelectric device is reversed. The main coolant supply then serves as a source of heat for the portion of coolant flow into the assembly, and both the heat extracted from the main coolant supply and the power input to the thermoelectric device are absorbed by the coolant for the inertial platform assembly.

Additional performance characteristics of thermoelectric devices may be found in Appendix 5A to Volume 2 of this report.

One advantage of thermoelectric devices is their high degree of reliability because of the lack of moving parts. Another advantage inherent in the system, as described above, is that it requires no separate means for producing coolant flow through the critical component. The main supply pumps are used to propel the coolant through all parts of the system. A disadvantage in using thermoelectric devices is the fact that they require a constant power input, even when no heat is pumped across the device. The cost of supplying this power would have to be compared with alternate means

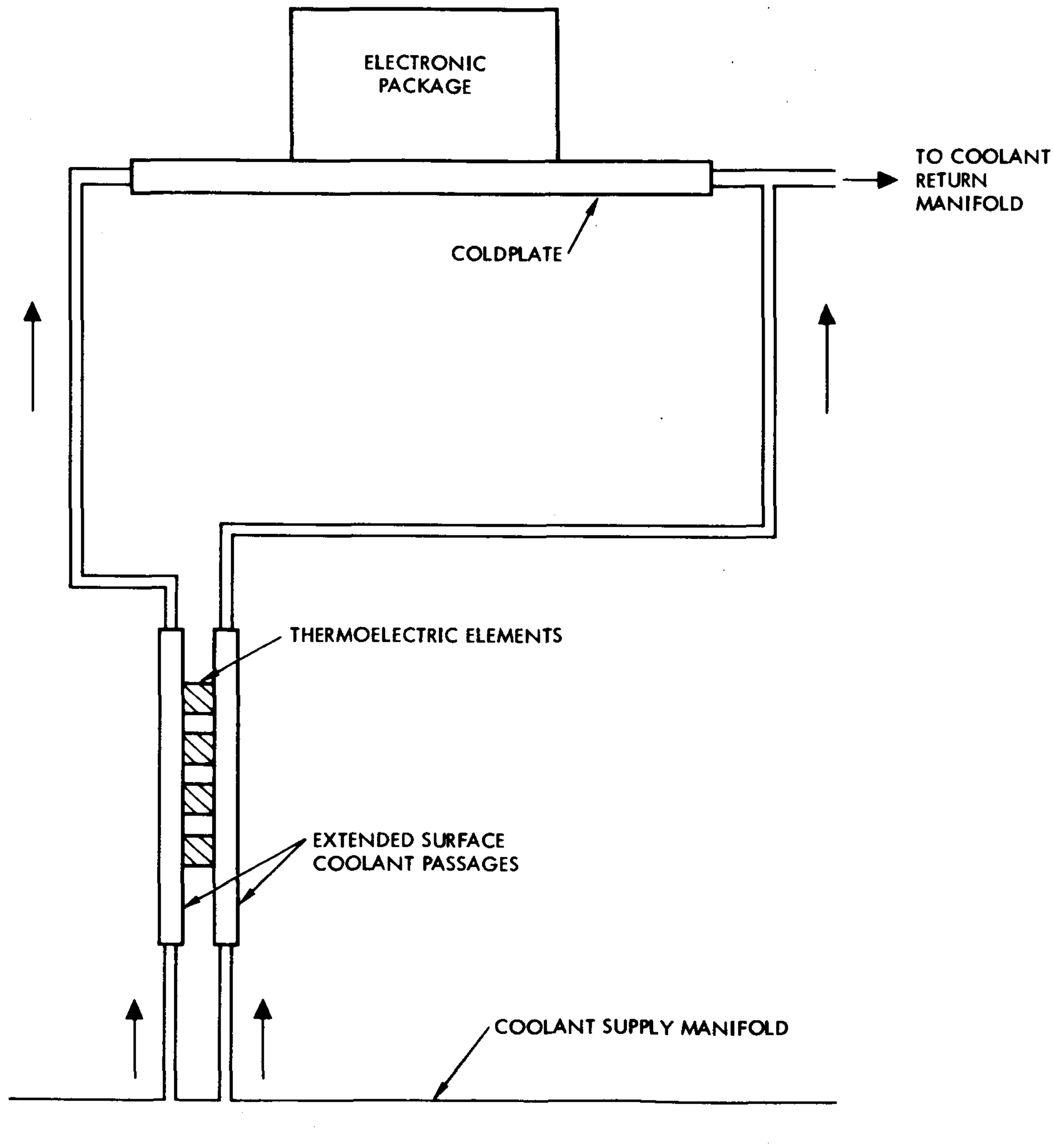

Figure 4-1. Thermoelectric Module Installation Schematic

for providing the required thermal control. To aid in this comparison, Figure 4-2 shows the performance of a typical thermoelectric device. Input power requirements are those for a 70-watt heat load, which is the power dissipation rate of the ST-124M inertial platform assembly. The performance characteristics shown indicate the rapid increase in input power requirements with the increase in the difference between hot junction and cold junction temperatures at constant hot junction temperature.

The significant approach to extending the Group I (including the accelerometer signal conditioner) lower temperature limit from 59 F to 30 F, (the lower limit of Group II) would be to provide a thermally actuated flow control valve at the outlet of each of the coldplates on which the Group I equipment is mounted. For the current equipment layout, this would include coldplates at IU locations 4, 5, and 20. Since only the Group I equipment is mounted on these coldplates, this appears to be an effective and convenient method.

For the internal or integrally cooled equipment, the ST-124M inertial platform assembly and the LVDC would require a thermally actuated flow control valve at the coolant outlet to limit the flow when the coolant temperature drops to the 30 F value.

Based on the foregoing discussion relating to thermal conditioning system modifications, recommended thermal conditioning system concepts for the missions postulated in Table 4-1 have been summarized in Table 4-4. Mission durations cover the range specified in Table 4-1, and electronic equipment heat loads are those shown in Figure 1-4 for specific mission duration. Use of a sublimator as a heat sink during the initial powered flight phase is recommended for all missions and durations. Flow control valves are suggested for critical items of equipment in instances where the listed electronic equipment heat load (full-on) results in coolant temperature at coldplate inlet below the minimum permissible for the equipment, based on NASA specification temperature limits. A space radiator is recommended whenever active heat removal is required beyond the initial powered flight phase of a mission. Table 4-4 indicates that such active heat removal is required only for mission durations in excess of 24 hours, for which the weight of an expendable heat sink becomes excessive.

Coolant heaters are recommended for mission durations that include periods of operation under standby conditions. Whether or not such heaters are necessary in every case depends on the lowest permissible coolant temperature for operating equipment and/or the lowest permissible storage temperature for nonoperating equipment. The ST-124M inertial platform assembly, for example, requires a minimum 50 F coolant temperature under operating conditions and a minimum 30 F temperature under nonoperating condition. Hence, if this item of equipment is turned on during standby

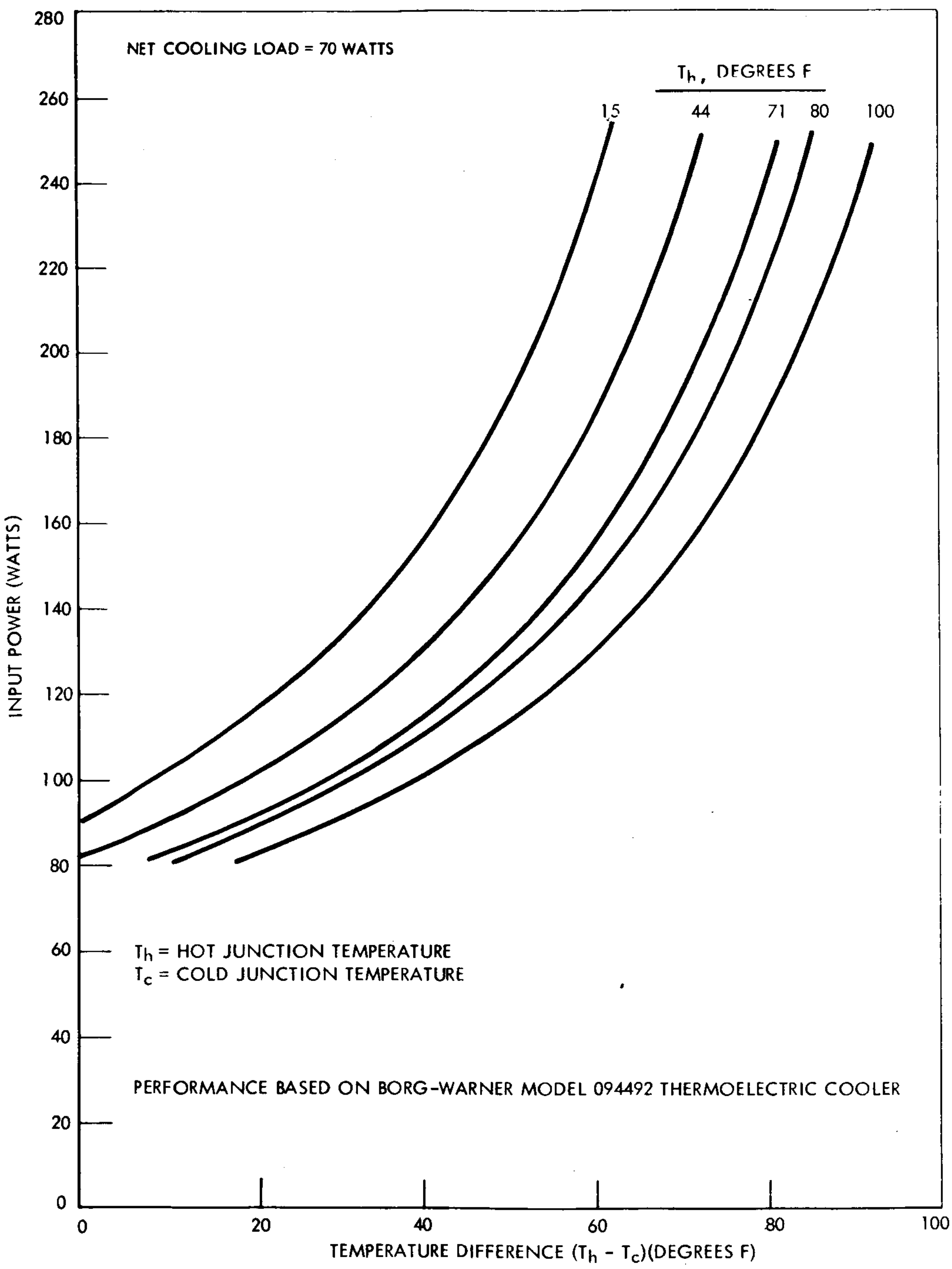

Figure 4-2. Thermoelectric Device Performance

Table 4-4. Recommended Thermal Conditioning Systems

Mission Profile			Electronic Equipment Heat Load (kw)		Physical Condition of Instrument Unit		Recommended Thermal Conditioning System
Orbit	Orientation	Duration	Full-On	Standby	Outer Shell	S-IVB Dome	
Near earth (200-n. mi)	X axis sun-oriented	10 hours	3.1	-	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Sublimator for initial powered flight Flow control valve for accelerometer signal conditioner
		24 hours	3.2	0.8	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Sublimator for initial powered flight Flow control valve for accelerometer signal conditioner (Coolant heaters to operate during standby condition)
		60 days	4.6	0.9	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Sublimator for initial powered flight (Coolant heaters to operate during standby condition)
		180 days	5.3	1.0	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Space radiator Sublimator for initial powered flight (Coolant heaters to operate during standby condition)
	Y axis sun-oriented	10 hours	3.1	-	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Sublimator for initial powered flight
		24 hours	3.2	0.8	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Sublimator for initial powered flight (Coolant heaters to operate during standby operation)
		60 days	4.6	0.9	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Space radiator Sublimator for initial powered flight (Coolant heaters to operate during standby operation)
		180 days	5.3	1.0	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Space radiator Sublimator for initial powered flight (Coolant heaters to operate during standby operation)
Synchronous	X axis sun-oriented	24 hours	3.2	0.8	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Sublimator for initial powered flight Coolant heaters for ST-124-M inertial platform assembly (and to operate during standby condition) Flow control valves for accelerometer signal conditioner, platform AC power supply, and ST-124 electronics
		60 days	4.6	0.9	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Sublimator for initial powered flight (Coolant heaters to operate during standby condition)
		180 days	5.3	1.0	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Sublimator for initial powered flight (Coolant heaters to operate during standby condition)
	Y axis sun-oriented	24 hours	3.2	0.8	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Sublimator for initial powered flight Flow control valve for accelerometer signal conditioner (Coolant heaters to operate during standby condition)
		60 days	4.6	0.9	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Sublimator for initial powered flight (Coolant heaters to operate during standby condition)
		180 days	5.3	1.0	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Space radiator Sublimator for initial powered flight (Coolant heaters to operate during standby condition)
Lunar mission (ground launch)	X axis sun-oriented	10 hours	3.1	-	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Sublimator for initial powered flight Coolant heater for ST-124-M inertial platform assembly Flow control valves for accelerometer signal conditioner, platform AC power supply, and ST-124 electronics
		24 hours	3.1	-	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Sublimator for initial powered flight Coolant heater for ST-124-M inertial platform assembly Flow control valves for accelerometer signal conditioner, platform AC power supply, and ST-124 electronics
	Y axis sun-oriented	10 hours	3.1	-	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Sublimator for initial powered flight Flow control valve for accelerometer signal conditioner
		24 hours	3.1	-	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Sublimator for initial powered flight Flow control valve for accelerometer signal conditioner
Lunar and deep space mission (orbital launch)	X axis sun-oriented	30 days	4.4	0.9	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Sublimator for initial powered flight (Coolant heaters to operate during standby condition)
		60 days	4.6	0.9	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Sublimator for initial powered flight (Coolant heaters to operate during standby condition)
	Y axis sun-oriented	30 days	4.4	0.9	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Space radiator Sublimator for initial powered flight and peak power loads (Coolant heaters to operate during standby condition)
		60 days	4.6	0.9	$\alpha_s/\epsilon = 0.20$	$\epsilon = 0.05$	Space radiator Sublimator for initial powered flight and peak power loads (Coolant heaters to operate during standby condition)

operation, heat must be supplied to the coolant because coolant equilibrium temperature will be below 50 F. If the inertial platform assembly has been supplied with a separate coolant temperature control system, such as a thermoelectric device, then the requirement for active heat addition to the main coolant supply will be determined by the temperature requirements of other equipment.

The applicability of the recommended TCS concepts rests upon the following assumptions with regard to a basic system and its properties:

1. The basic system consists of 16 coldplates through which a water-methanol coolant is circulated.

2. A sublimator or similar device is available for reducing the temperature of the coolant through heat transfer to an expendable heat sink (stored water).

3. Means are provided for separate control of critical electronic packages so that the resulting permissible coolant temperature range is 30 F to 80 F. This modification to the present IU ECS is assumed to be incorporated for missions in excess of 10 hours.

4. The coolant flow rate is a nominal 60 pounds per hour per cold coldplate.

5. The infrared emissivities of the coldplates and the electronic packages mounted on them are 0.18 and 0.90, respectively.

6. A value of 0.18 is preferable for the solar absorptivity of the IU outer skin to minimize the varying effects of solar radiation as a function of orbital position. However, when the IU is in a minimum solar heating orientation, higher values of α_s may be desirable under conditions of low equipment power dissipation levels. Therefore, the value of α_s to be used is not unique but will depend on orbital altitude, vehicle orientation, and the level of equipment power dissipation.

7. A value of 0.9 is desirable for the infrared emissivity of the IU outer skin when the IU is in a maximum solar heating orientation and the equipment power dissipation level is high. For other orientations and equipment power dissipation levels, this emissivity value may result in coolant temperatures that are too low. Therefore, the value of IU outer skin emissivity to be used will have to be determined on the basis of the amount of absorbed incident radiation and equipment power dissipation level.

8. The emissivity of the S-IVB dome is assumed to vary between 0.05 and 0.9, depending on the dome temperature profile and the desired effect on the IU temperature level.

Based on these assumptions, the recommended control system concepts may be applied to the postulated missions on the basis of required operating time and equipment power dissipation level. The relationship between these variables is summarized in Table 4-4. It is to be noted that the suitability of the recommended concepts is generally insensitive to mission profile, but depends primarily on mission duration and equipment power dissipation profile.

Operating times up to 10 hours have been assumed to be applicable only to the 200-nautical-mile earth orbital mission and to the lunar mission with ground launch. For these missions, constant levels of equipment power dissipation have been postulated. Any cooling requirement which results can be met most conveniently by the use of an expendable coolant (water) on a demand basis. A possible alternate means of heat rejection exists in terms of the relatively low temperature of the S-IVB dome, which may be of use in IU thermal control through selection of an appropriate value of dome infrared emissivity.

Operating times up to 24 hours have been assumed to be applicable to all postulated missions except the lunar and deep space mission with orbital launch. For this duration, the recommended primary heat sink is still the present expendable coolant system, although different equipment power dissipation profiles are considered possible. An alternate source of heat sink capacity is hydrogen vent gas from the S-IVB stage, which could be used on a demand basis.

For operating times up to 60 days, which have been assumed to be applicable to all postulated missions except the lunar mission with ground launch, hydrogen vent gas from the S-IVB stage cannot be relied upon as a possible alternate heat sink because it is considered to be unavailable after the first 24 hours. Also, the use of an expendable heat sink over this length of time will result in prohibitive weight penalties. Therefore, the recommended primary heat sink is a space radiator, if the equipment power dissipation level is such that continuous heat rejection is required. If the equipment power dissipation is such that passive control is adequate, the space radiator and/or an expendable coolant could serve to provide heat sink capacity during peak power periods. An alternate source of heat sink capacity is considered to be the expendable coolant system if water is available as a fuel cell by-product.

For operating times in excess of 60 days, which have been assumed to be applicable only to the two earth orbital missions, it is assumed that a power source other than fuel cells will be employed. Hence, the heat sink capacity of by-product water will not be available, and only the primary concept (space radiator) can be considered applicable. A limited quantity of expendable coolant could be employed during peak power periods if the thermal lag in the system is insufficient.

The primary and alternate heat sinks are listed in Table 4-5 for the four assumed operating time periods.

One alternate source of available heat capacity indicated in Table 4-5 is the hydrogen vent gas from the S-IVB stage. The sensible heat capacity of this cryogenic gas could be used to absorb heat rejected from the environmental control system when such heat rejection is required. However, the availability of hydrogen vent gas is a function of S-IVB propellant utilization and may not be predictable with sufficient certainty. Therefore, the use of hydrogen vent gas as a heat sink is recommended only as an alternative of less than prime desirability.

Table 4-5. Recommended Heat Sinks

Operating Time	Heat Sinks	
	Primary	Alternate
Up to 10 hours	Expendable coolant: Stored water on demand basis	S-IVB Dome
Up to 24 hours	Expendable coolant: Stored water on demand basis	Expendable coolant: S-IVB hydrogen vent gas on demand basis
Up to 60 days	Space radiator plus expendable coolant: Stored water on demand basis	Expendable coolant: Fuel cell by-product water
Over 60 days	Space radiator plus expendable coolant: Stored water on demand basis	None

NOTE: Electrical heaters or another heat source may be required on a demand basis during a portion of the ascent and orbital phases. The duration is dictated by the S-IVB dome temperature. Radiation from IU outer skin to deep space heat sink is utilized for all operating times.

In the thermal analysis section of the report it was shown that thermal control of the IU is possible under zero coolant flow conditions. To take advantage of this condition, it would be necessary to provide separate thermal control for the critical items of equipment mentioned earlier. In addition to providing this equipment with thermoelectric devices or other active systems, it would be necessary to have a completely separate coolant circuit for this equipment to have a source of coolant flow. It should be apparent that a system designed to operate with no coolant flow to most of the coldplates will have compensating features, such as added complexity, which might be less desirable than those of other systems. Hence, the zero coolant flow system is recommended only as an alternative, especially for the cases of loss of coolant through leakage and loss of flow through pump failure.

CONCLUSIONS AND RECOMMENDATIONS

As a result of the study, several conclusions can be made with regard to extending the operational life and decreasing the mission sensitivity of the current ECS. Based on possible modifications to the current system, recommended system concepts have been made. The conclusions, recommended concepts, and recommendations for further investigation are as follows:

For the various orbits, vehicle orientations, and astrionic equipment considered in this study, the operational life of the current ECS can be extended, with modifications, as long as the equipment power dissipation rate is within limits for which no heat rejection or addition is required. The minimum modifications for the current configuration of 16 coldplates and integrally cooled equipment with coolant flow consists of (1) maximizing the heat rejection capability of the coldplate and the IU outer skin surface by selection of proper surface coatings (Table 4-4) and (2) extending the tolerable coolant temperature range by providing individual thermal control for each of the few critical pieces of equipment. These modifications can be readily incorporated into the present configuration. Once thermal equilibrium has been established following powered flight, there is a range of power dissipation levels within which the three categories of IU equipment noted may be operated in a passive mode (no active heating or cooling of the circulating fluid in the coldplates) for an indefinite period. Within these power dissipation levels, the IU thermal control system design is insensitive to mission duration.

Based on the minimum modifications indicated in the previous paragraph, there is a narrower range of total IU equipment operating power dissipation levels over which the IU thermal control system design is insensitive to the earth orbital mission change. Within this range of power dissipation levels, the IU may be operated on changed earth orbital missions without a redesign of the thermal control system.

Based on the minimum modifications indicated in the first paragraph, there is a range of total IU equipment operating power dissipation values (much narrower than those indicated in the second paragraph) over which the IU thermal control system design is insensitive to the missions and orbits considered in this study. This possibility of a single design for all missions has been indicated, but the power dissipation range may be too narrow to be practical.

The effect of vehicle axial orientation on the allowable power dissipation levels is sensitive to the α_s/ϵ of the IU outer skin. For a α_s/ϵ of 0.20, the allowable power dissipation levels vary moderately between the axial orientation for maximum solar heat and minimum solar heat. For a α_s/ϵ approaching the value of 1.0, the allowable power dissipation levels should vary considerably for different vehicle orientation. In addition, for orbital conditions that have earth shadow periods, α_s/ϵ, as it approaches the value of 1.0, results in a wide swing from the maximum to minimum net heat load on thermal conditioning systems on a per-orbit basis. Thus, it would be desirable to select a value for the α_s/ϵ ratio to minimize or dampen the variation in the net heat load (near the 0.2 value).

For the short-duration mission (less than 24 hours), it appears possible that the S-IVB dome could be used as the heat sink by selection of the proper dome surface properties.

The results of the study for the S-IVB dome emissivity approaching 0.90 indicate that heat addition appears necessary to maintain the coolant temperature above the minimum allowable value. Thus, a lower value for the S-IVB dome emissivity should minimize the requirements for heating or cooling. The selection of the emissivity value will depend upon the S-IVB dome temperature-time profile and expected heat load.

For an emergency mode, with coolant flow, a change in vehicle orientation from maximum to minimum solar heat (or vice versa) may provide some benefit, depending on whether cooling or heating is required.

A purely passive approach (no coolant flow) may be possible for the case of low power dissipation and with provisions for individual thermal control for the critical components. This approach should be investigated further, particularly since it may be used during an emergency due to coolant circulation failure resulting from pump failure or complete loss of coolant.

For the short-duration peak or spike equipment power dissipation, the system thermal mass may be sufficient to absorb the transient heat load. A brief analysis indicates that, for a peak equipment heat load of less than two hours, the system thermal mass appears sufficient. As the duration of the peak or spike heat load increases and approaches ten hours, the system mass is not sufficient and so heat rejection by an appropriate means must be used.

For the missions and postulated heat load profiles, various system concepts and alternatives have been presented. For the short-duration missions (less than 24 hours), one possible approach would be to utilize the thermal capacity of the S-IVB dome to provide the necessary heat sink, thus avoiding additional heat rejection provisions. As the mission time increases and after the S-IVB dome temperature reaches an assumed equilibrium level condition (60 F), one approach would be to incorporate a space radiator, particularly for a sustained heat load condition above the power dissipation level, which requires no active cooling of the circulating fluid. Thus, as the mission time increases and the heat load remains at a relatively high level, requiring heat rejection, the system complexity increases.

For the heat load profile that has peak or spike heat loads, the recommended approach has been based on the duration of the peak or spike heat load.

The thermal analysis performed during this study has provided the necessary data from which to establish the system concepts applicable to the type of missions described in this report. There are several areas for which a more detailed thermal analysis should be performed, using a more refined or detailed thermal model than the one used in this study. From a reliability standpoint, a no-flow condition, a purely passive method, should be investigated further to simulate the condition of pump failure or loss of coolant due to leakage in the coolant lines or coldplates. Failure of this type may be critical for only a few of the electronic packages, so the overall instrument unit may be able to function on a degraded basis. Another area for further investigation is the heat transfer between the S-IVB dome and the thermal conditioning system for various combinations of S-IVB dome temperature profiles and dome surface properties. The analysis presented in this report suggests the possibility of utilizing the S-IVB dome as a heat sink for the short-duration missions.

A better understanding of the system thermal lag is required if it is to be fully utilized for transient or peak equipment power dissipation cases. In addition, the advantages of using heat storage materials (particularly those with high heat of fusion at or near the system temperature) should be investigated as a means of dampening the effects of variation in orbital heat loads.

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APPENDIX

SUMMARY OF HEAT BALANCE ANALYSIS

The results of the heat balance analysis for the active thermal control system are summarized in Figures A-1 through A-23 for both the initial orbits (0 to 35 hours of orbit time) and the condition of S-IVB dome temperature equilibrium. For the latter, the IU heat load shown in the graphs represents the average heat load rate over a complete orbit. In many instances, the variation in heat load as a function of orbital position was found to be relatively large, especially in those cases in which the environmental conditions are subject to wide variation due to a high value of α_s (solar absorptivity) for the IU outer shell. This is evident from a comparison of Figures A-24 and A-25, which represent data for the same orbital condition, vehicle orientation, coolant coldplate inlet temperature (50 F), and electrical equipment heat load (150 watts per coldplate). An increase in the value of solar absorptivity from 0.18 to 0.90 results not only in an increase of more than threefold IU heat load, but also in departure of about ±3000 Btu's per hour from the average value during one complete orbit. With the lower value of α_s, it is limited to about ±650 Btu's per hour, as shown in Figure A-24. The IU outer shell emissivity value of 0.90 was used in all cases.

The variation in IU heat load as a function of coolant coldplate inlet temperature is shown in Figures A-1, A-2, and A-3 for the ten orbital conditions described in Table A-1. The data presented are for constant electrical equipment heat loads of 1.51, 3.11, and 4.71 kilowatts, respectively, corresponding to individual coldplate heat loads of 50, 150, and 250 watts; a solar absorptivity (α_s) value of 0.18 for the IU outer shell is applicable to all three figures. A net heat gain (+) in this and subsequent figures indicates the amount of heat that must be absorbed by a heat rejection system, while a net heat loss (-) is the amount of heat that must be added to the system to maintain the assumed coolant temperature. Thus, for Case 9 in Figure A-2, 1250 Btu's per hour must be rejected from the thermal control system to maintain the coolant coldplate inlet temperature at 30 F. Likewise, no heat rejection or addition is required for this case when a coolant coldplate inlet temperature of 42 F is used. In connection with the data in Figure A-2, the differences shown among the results for some of the cases are not significant. Case 4, for example, is identical to Case 6. Hence, heat loads for these cases may be expected to be the same. However, calculated results show a difference between the two cases, reflecting the lack of complete accuracy in the calculation method. In this method, the IU heat load was obtained by the summation of individual coolant

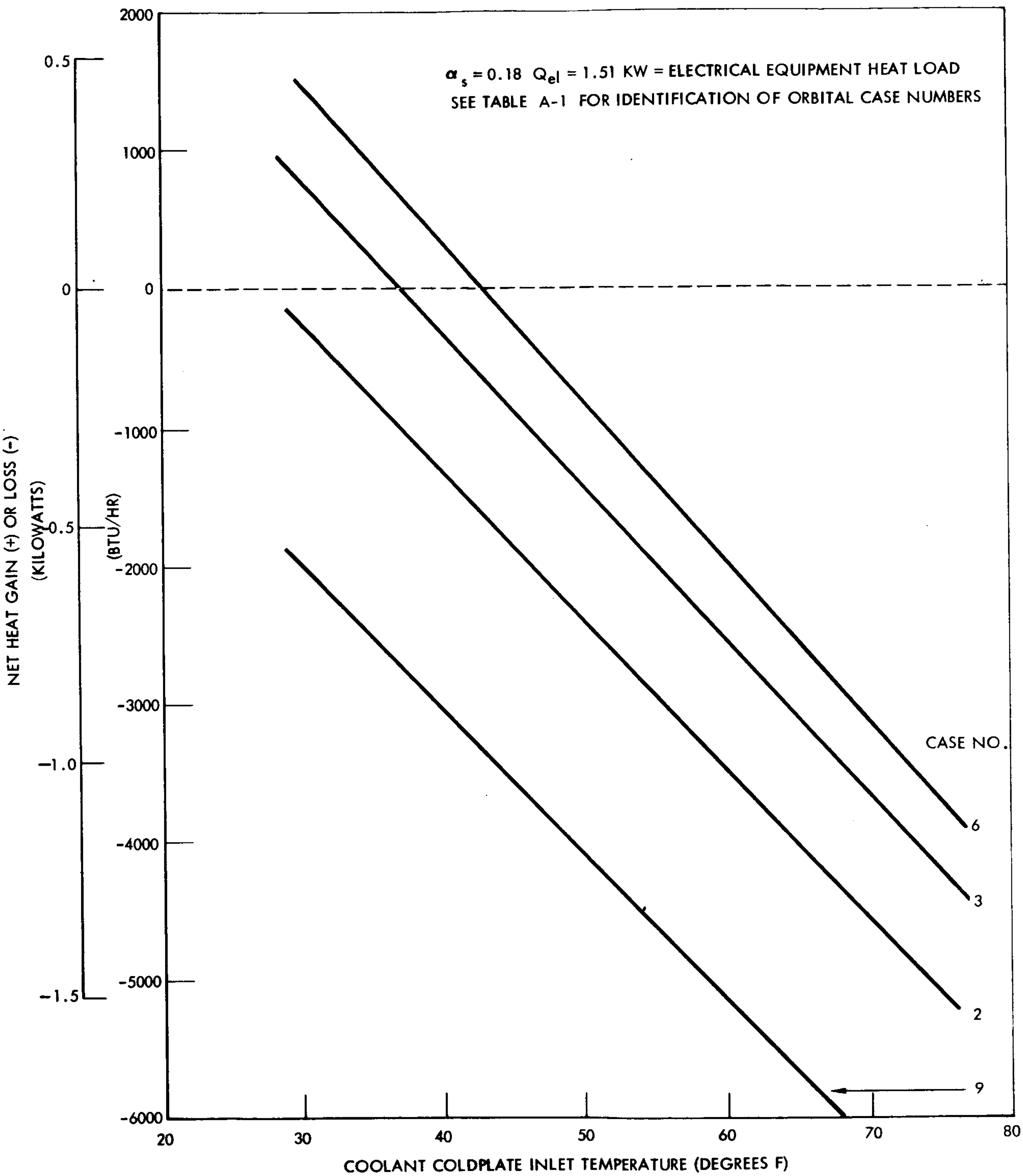

Figure A-1. Instrument Unit New Heat Gain or Loss Versus Coolant Coldplate Inlet Temperature, α_s = 0.18, Q_{el} = 1.51 kw

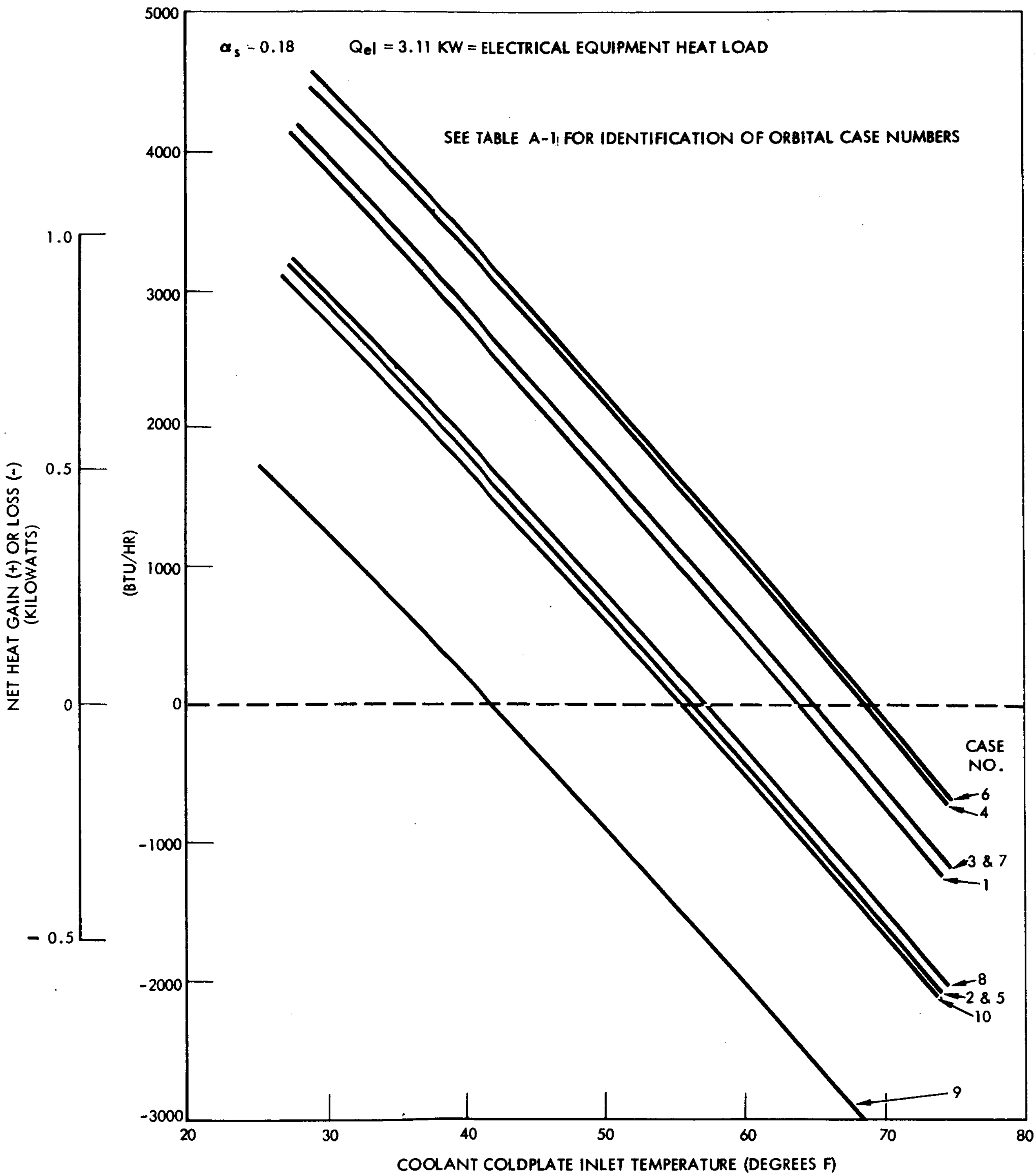

Figure A-2. Instrument Unit Net Heat Gain or Loss Versus Coolant Coldplate Inlet Temperature, α_s = 0.18, Q_{el} = 3.11 kw

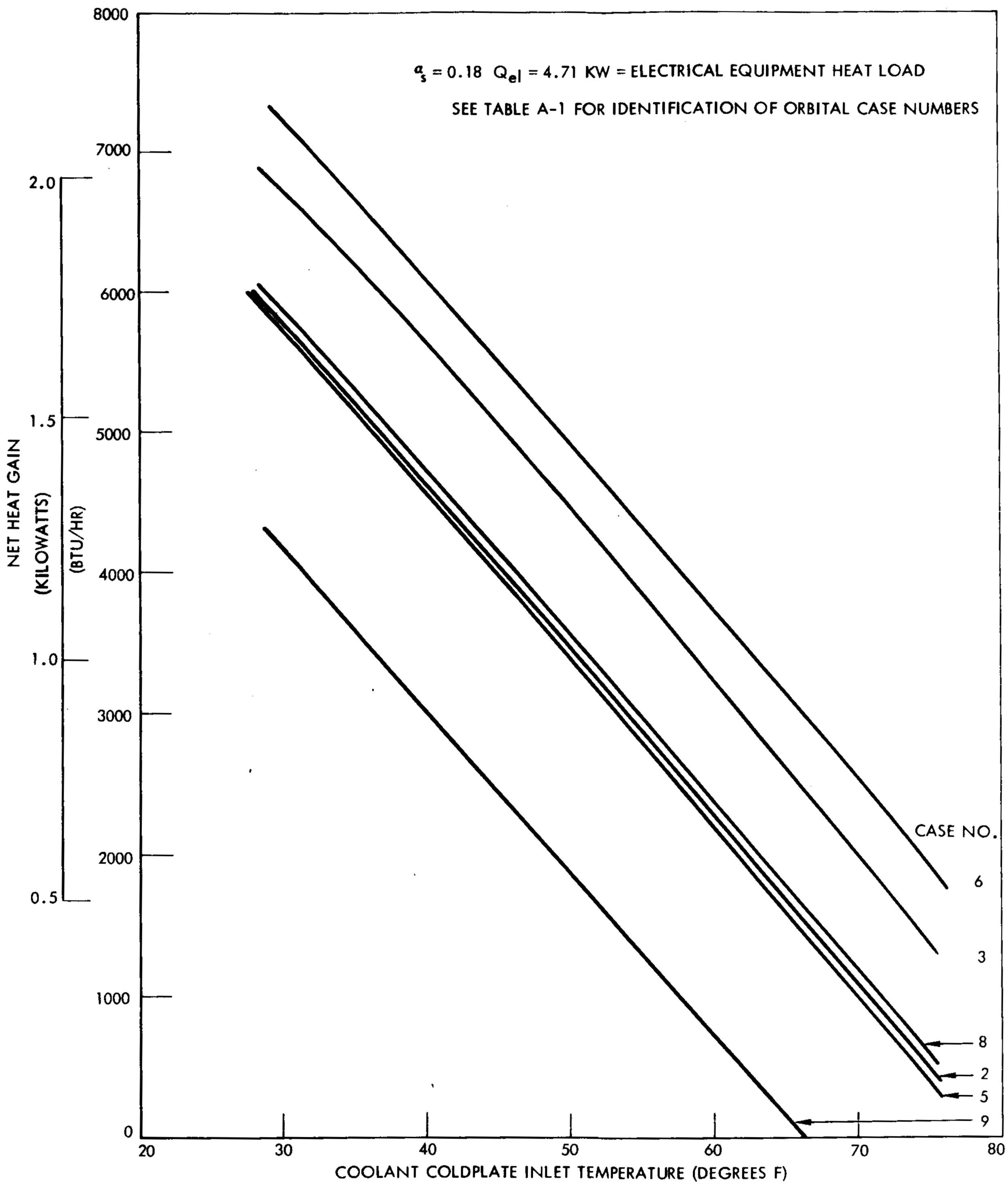

Figure A-3. Instrument Unit Net Heat Gain Versus Coolant Coldplate Inlet Temperature, $\alpha_s = 0.18$, $Q_{el} = 4.71$ kw

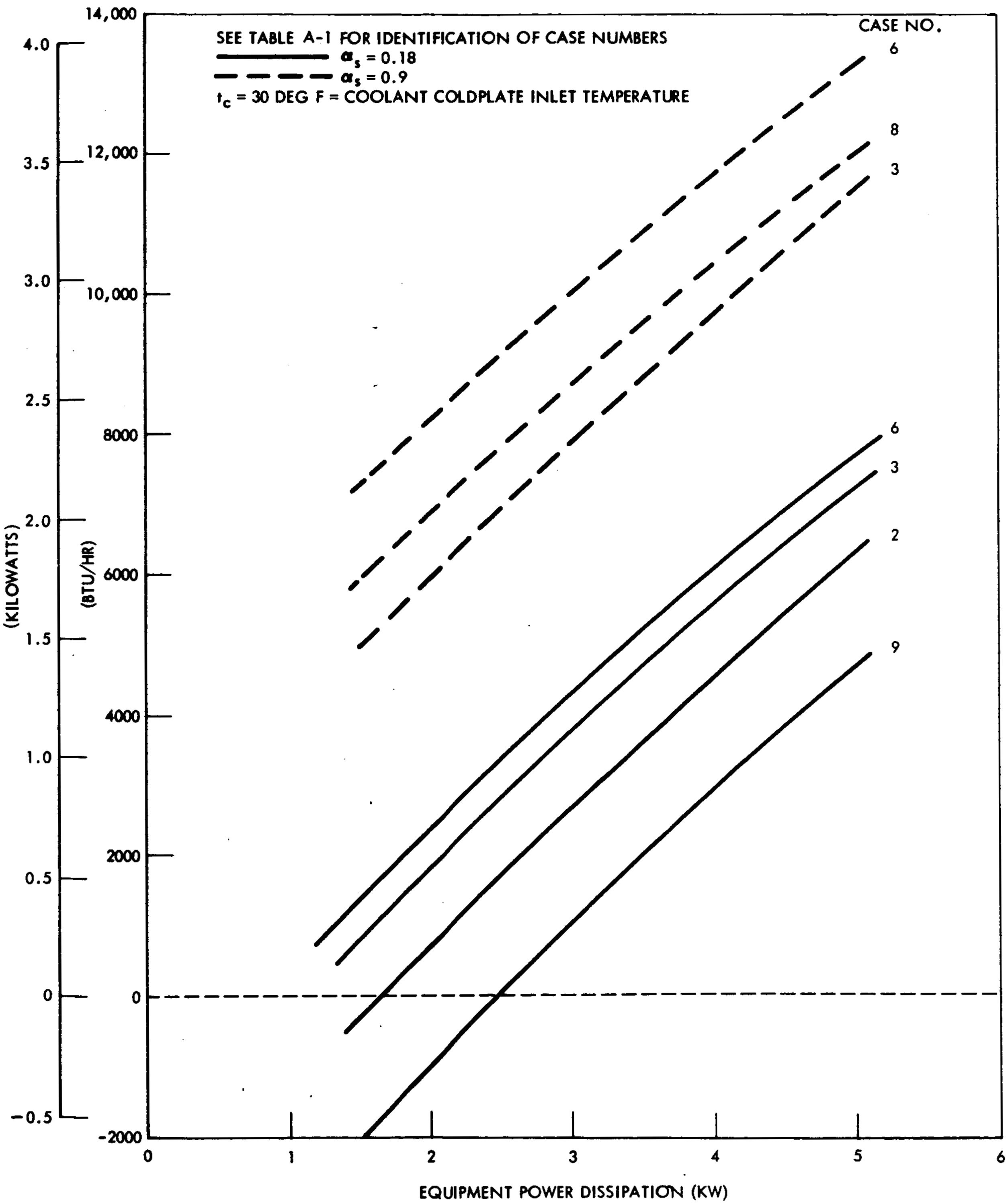

Figure A-4. Instrument Unit Net Heat Gain or Loss Versus Equipment Power Dissipation, t_c = 30 F

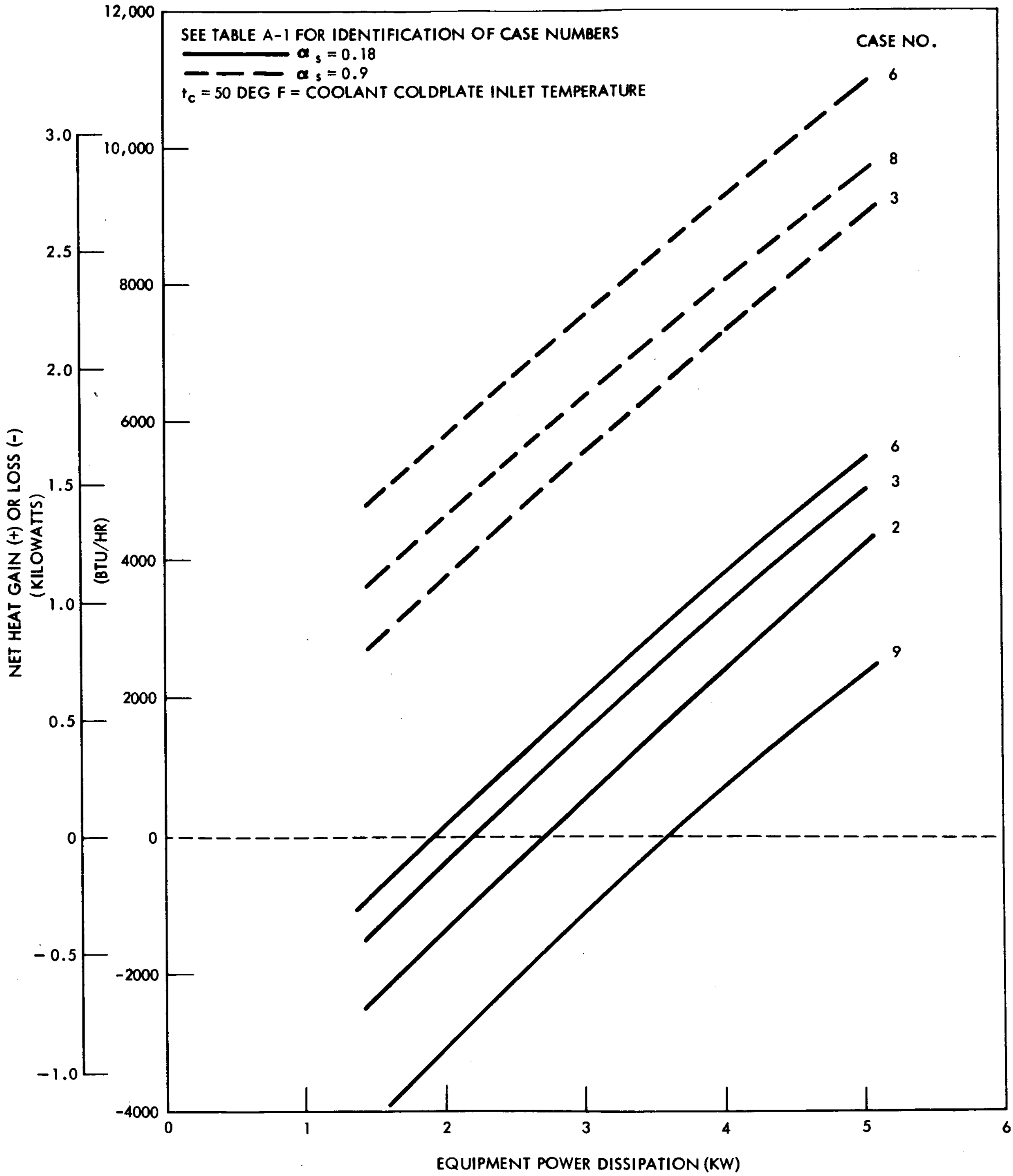

Figure A-5. Instrument Unit Net Heat Gain or Loss Versus Equipment Power Dissipation, t_c = 50 F

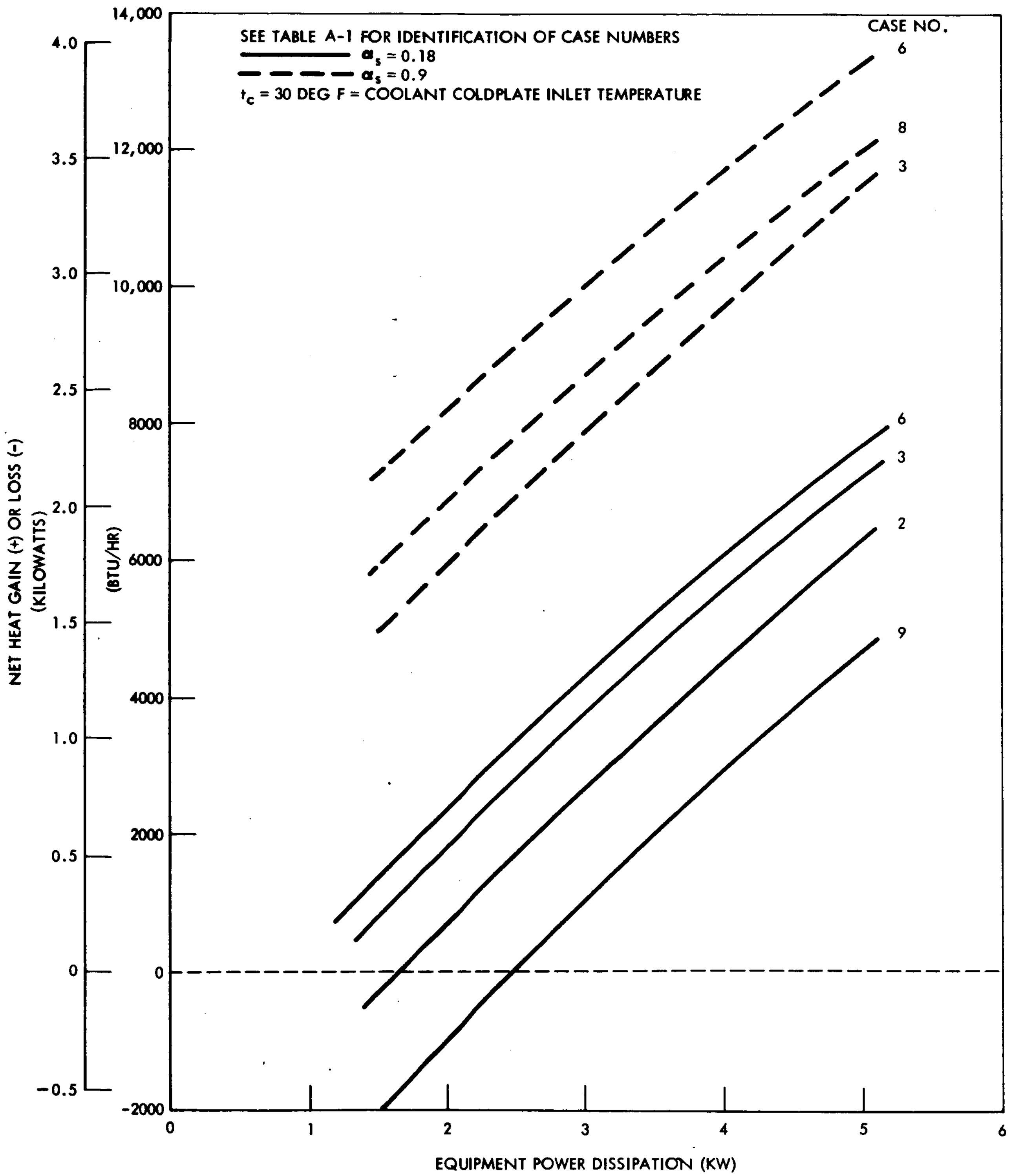

Figure A-4. Instrument Unit Net Heat Gain or Loss Versus Equipment Power Dissipation, t_c = 30 F

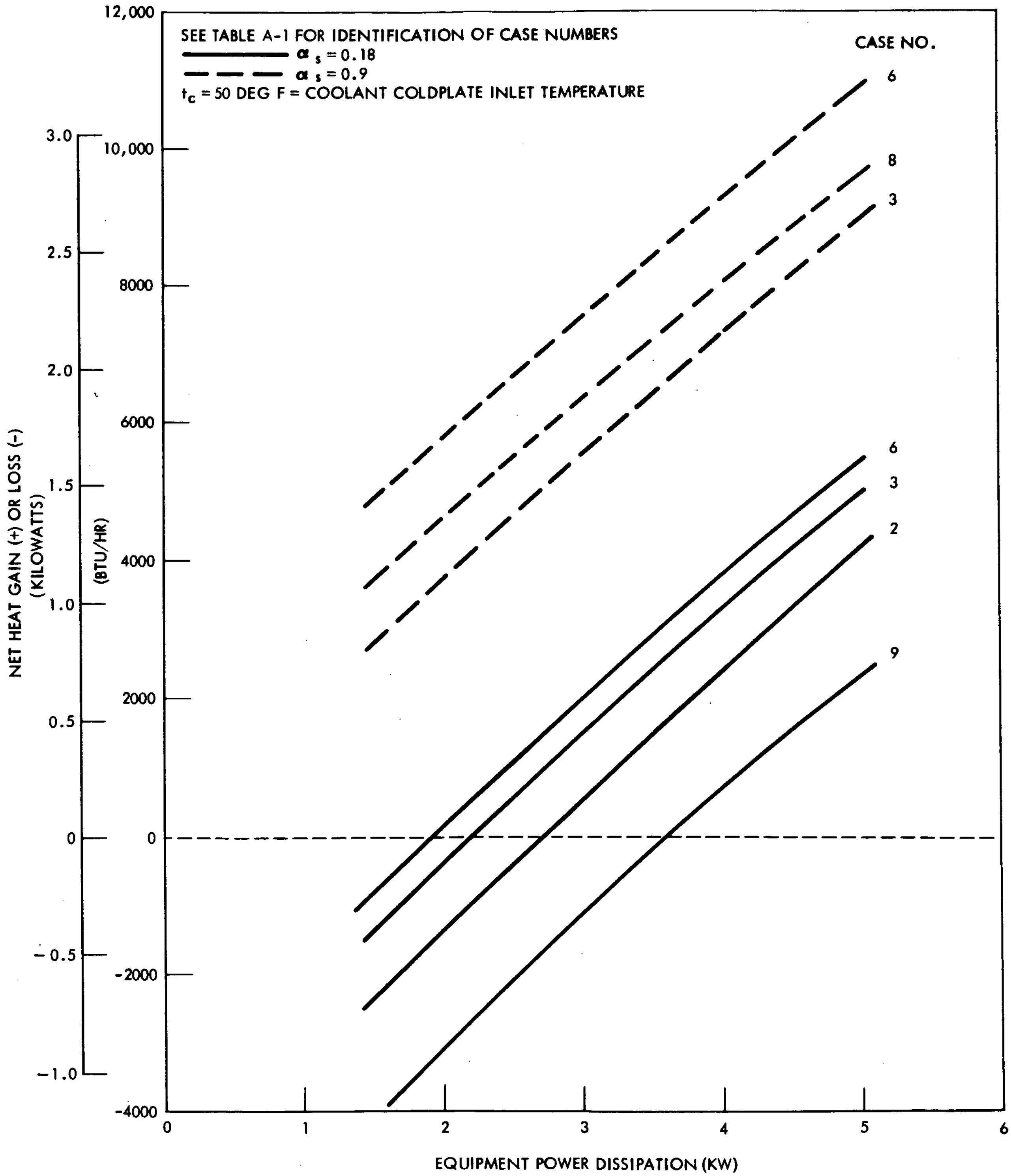

Figure A-5. Instrument Unit Net Heat Gain or Loss Versus Equipment Power Dissipation, t_c = 50 F

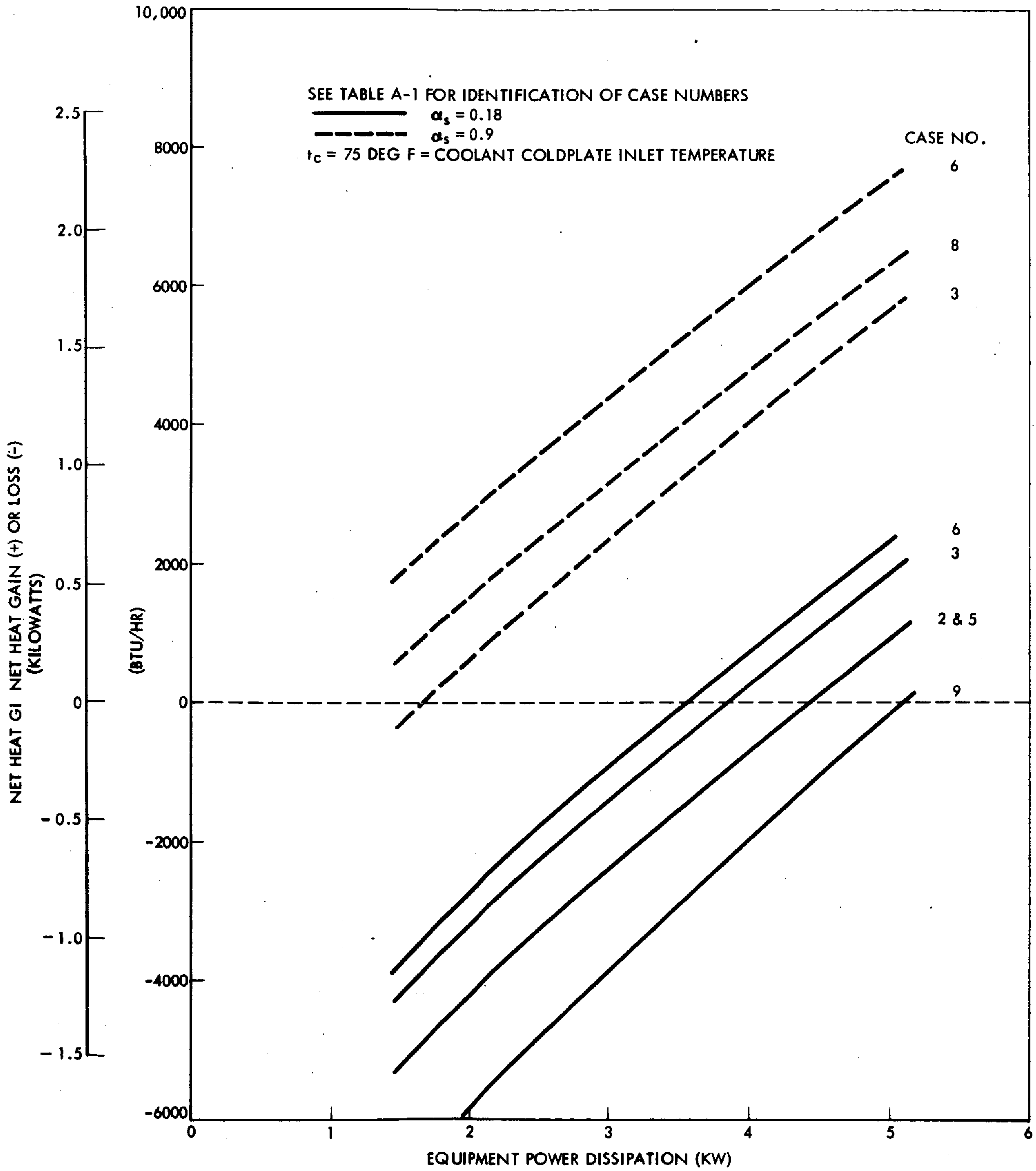

Figure A-6. Instrument Unit Net Heat Gain or Loss Versus Equipment Power Dissipation, t_c = 75 F

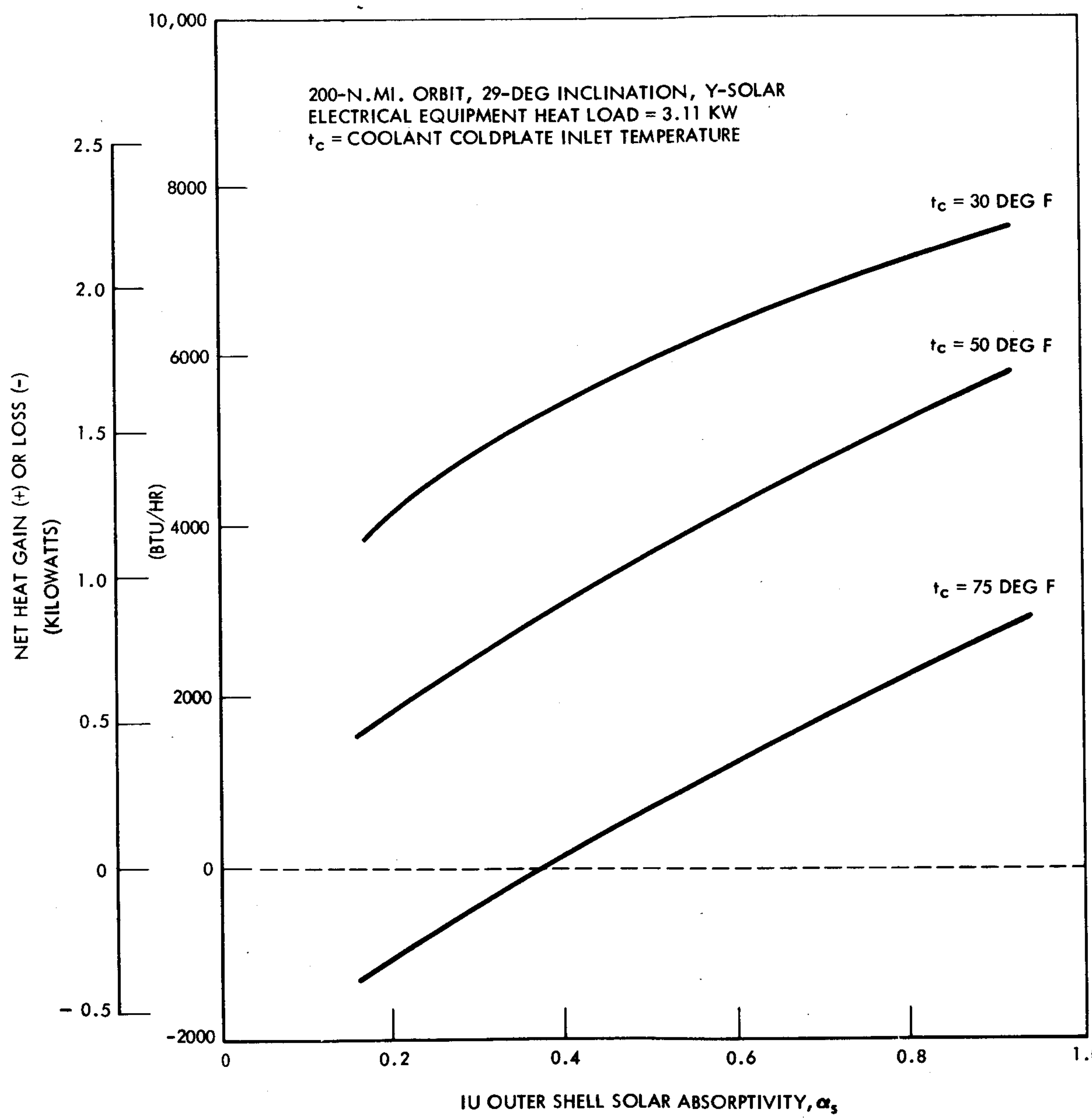

Figure A-7. Instrument Unit Net Heat Gain or Loss Versus IU Outer Shell Absorptivity (α_s), 200-N. Mi. Orbit, Q_{el} = 3.11 kw

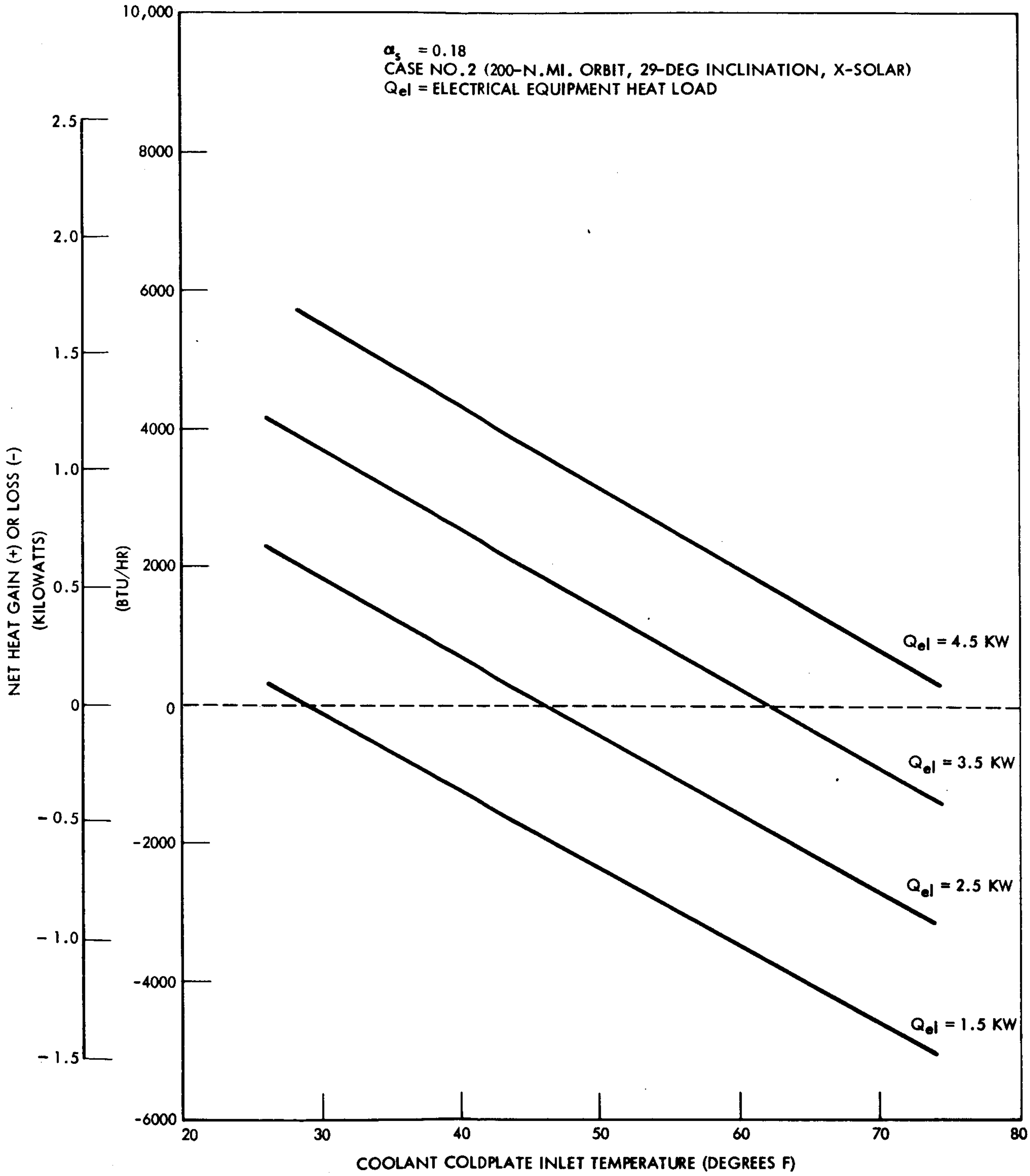

Figure A-8. Instrument Unit Net Heat Gain or Loss Versus Coolant Coldplate Inlet Temperature, Case No. 2

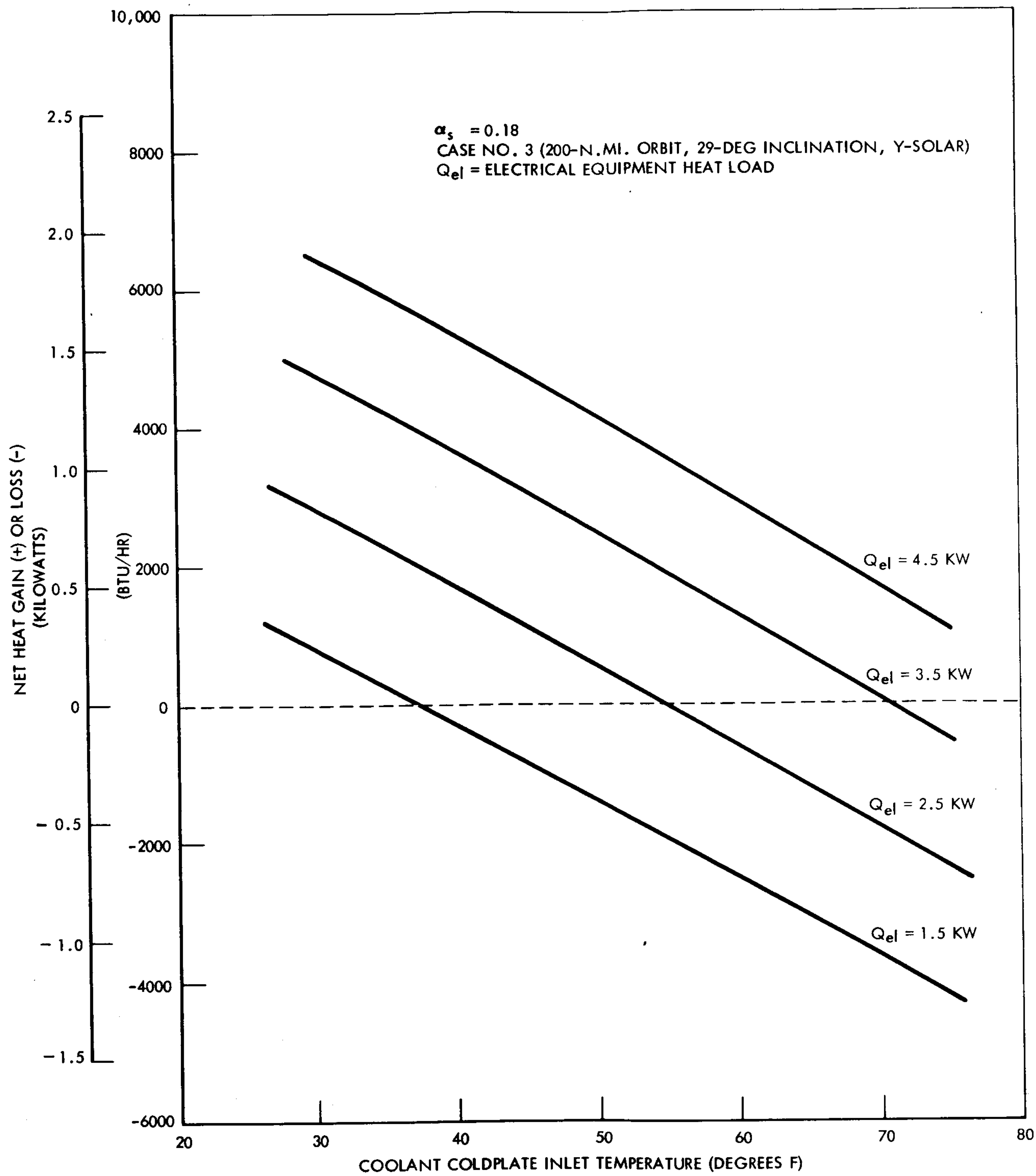

Figure A-9. Instrument Unit Net Heat Gain or Loss Versus Coolant Coldplate Inlet Temperature, Case No. 3

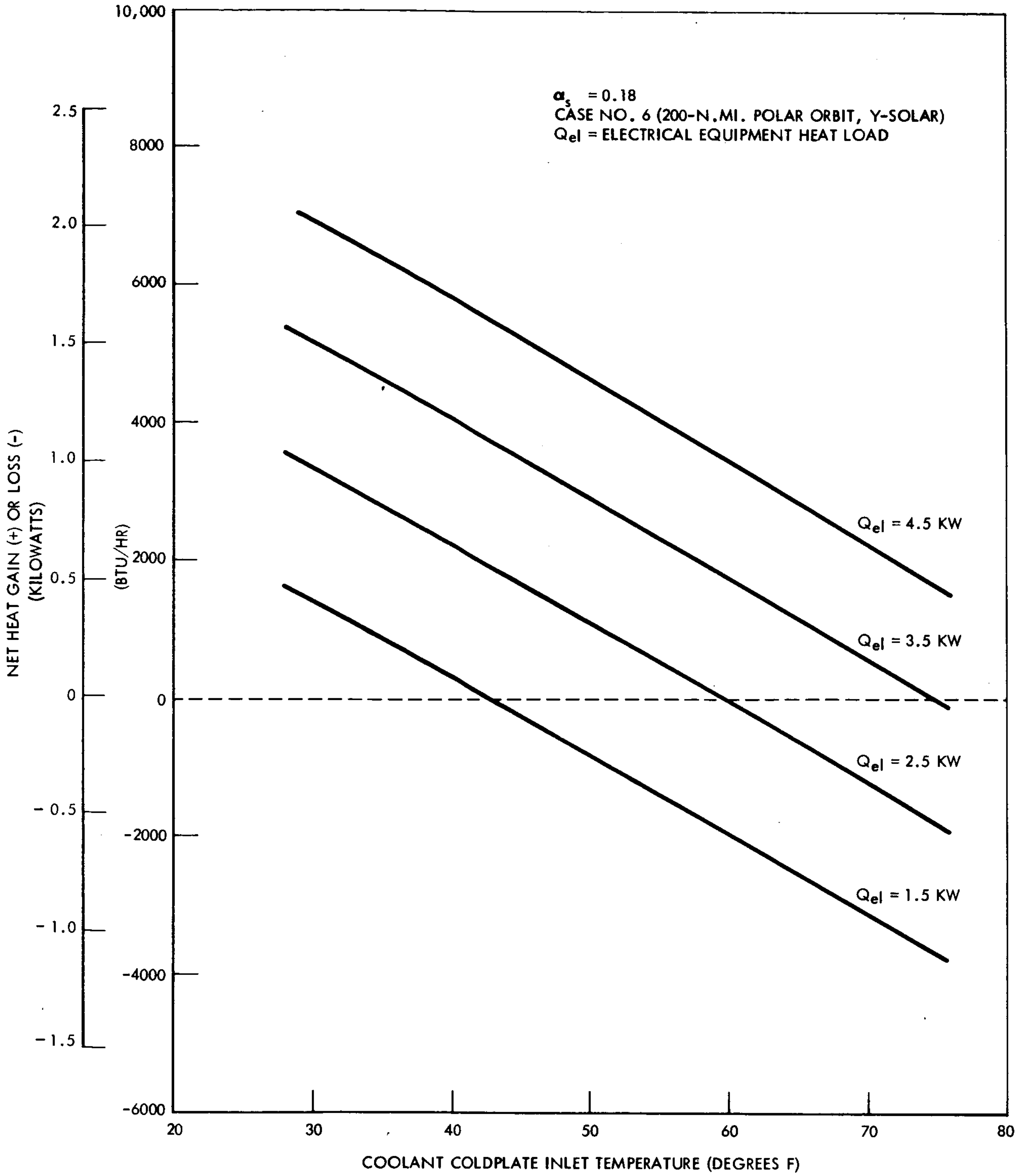

Figure A-10. Instrument Unit Net Heat Gain or Loss Versus Coolant Coldplate Inlet Temperature, Case No. 6

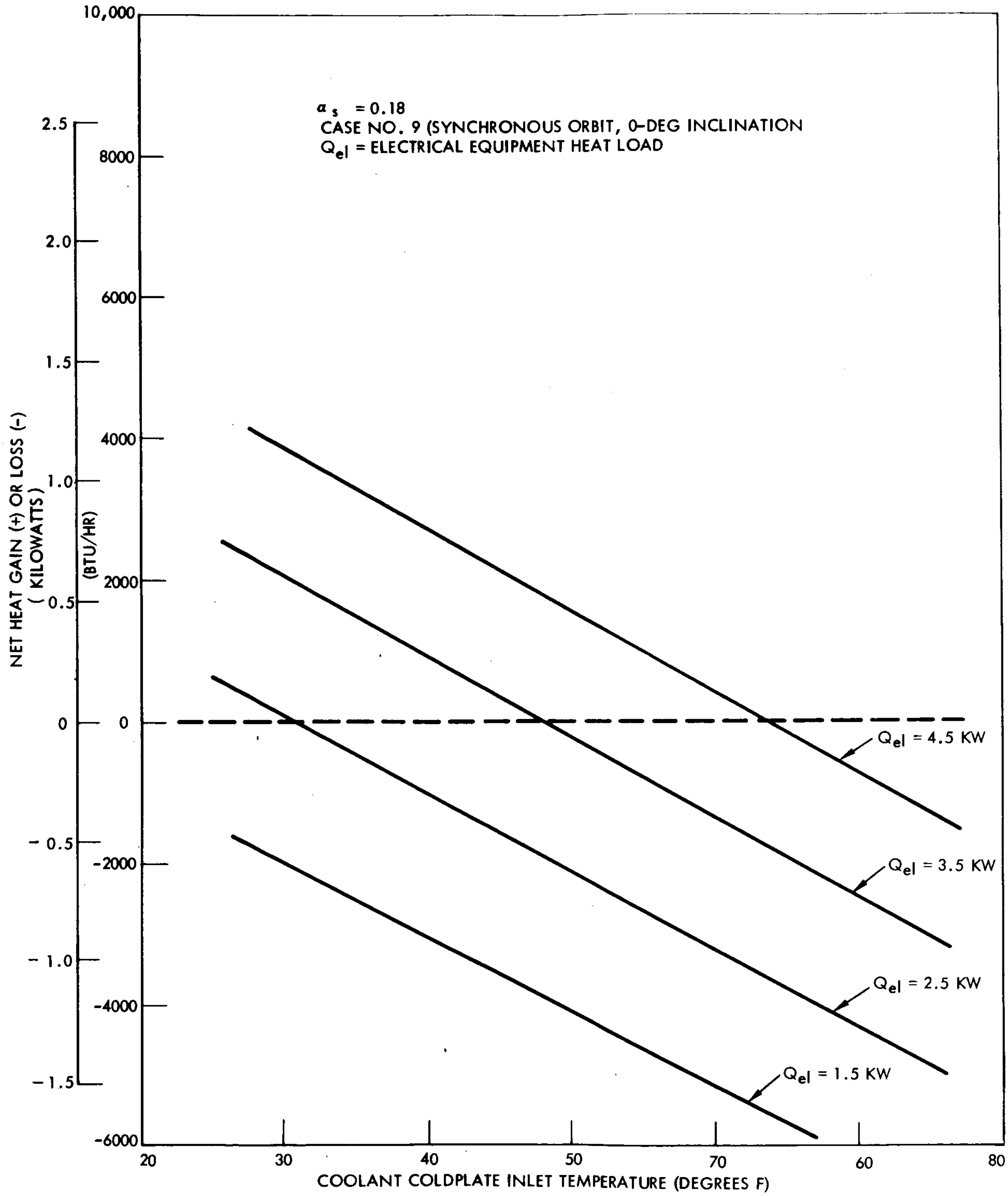

Figure A-11. Instrument Unit Net Heat Gain or Loss Versus Coolant Coldplate Inlet Temperature, Case No. 9

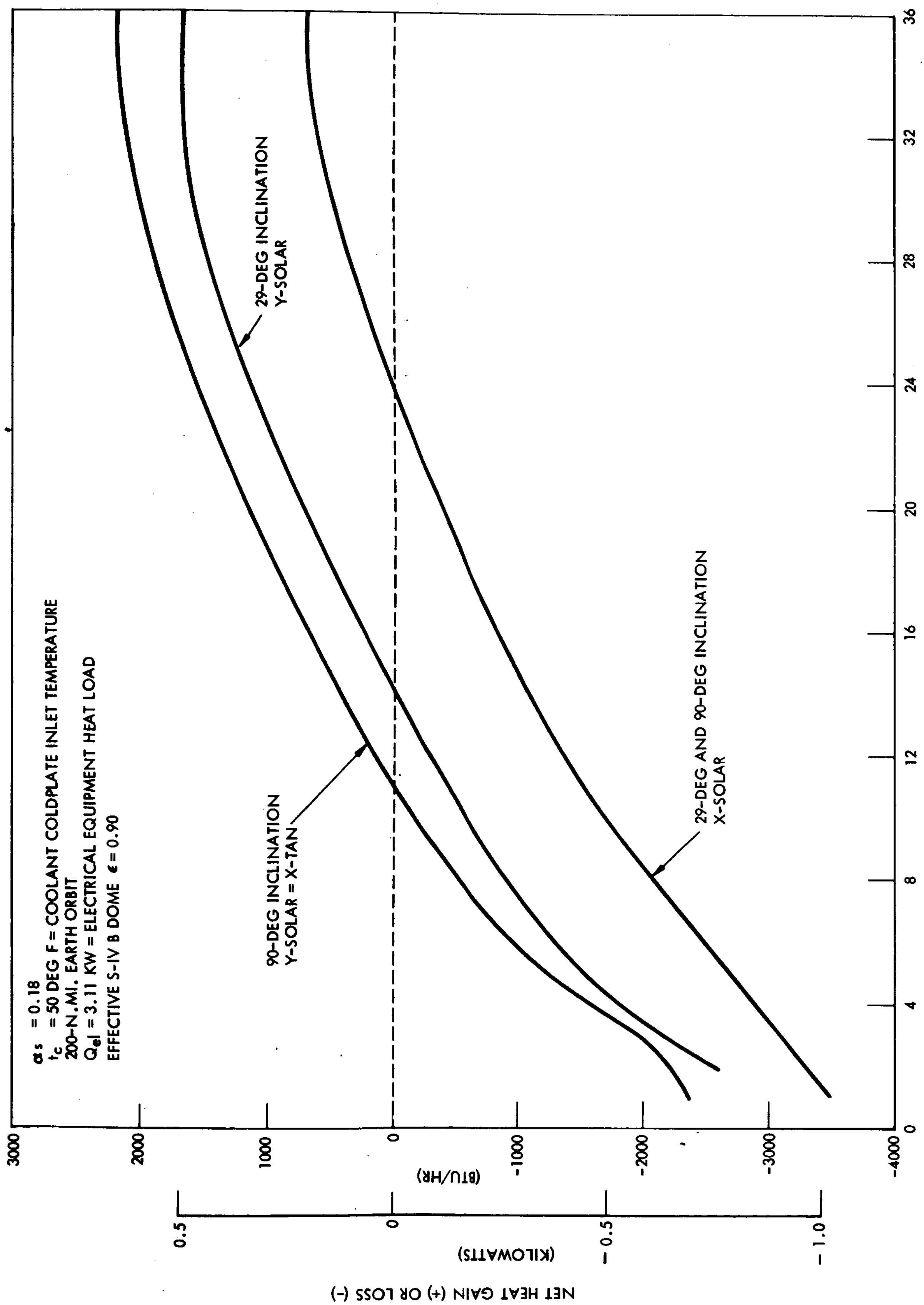

Figure A-12. Instrument Unit Net Heat Gain or Loss Versus Orbit Time, 200-N. Mi. Orbit, t_c = 50 F, Q_{el} = 3.11 kw, α_s = 0.18

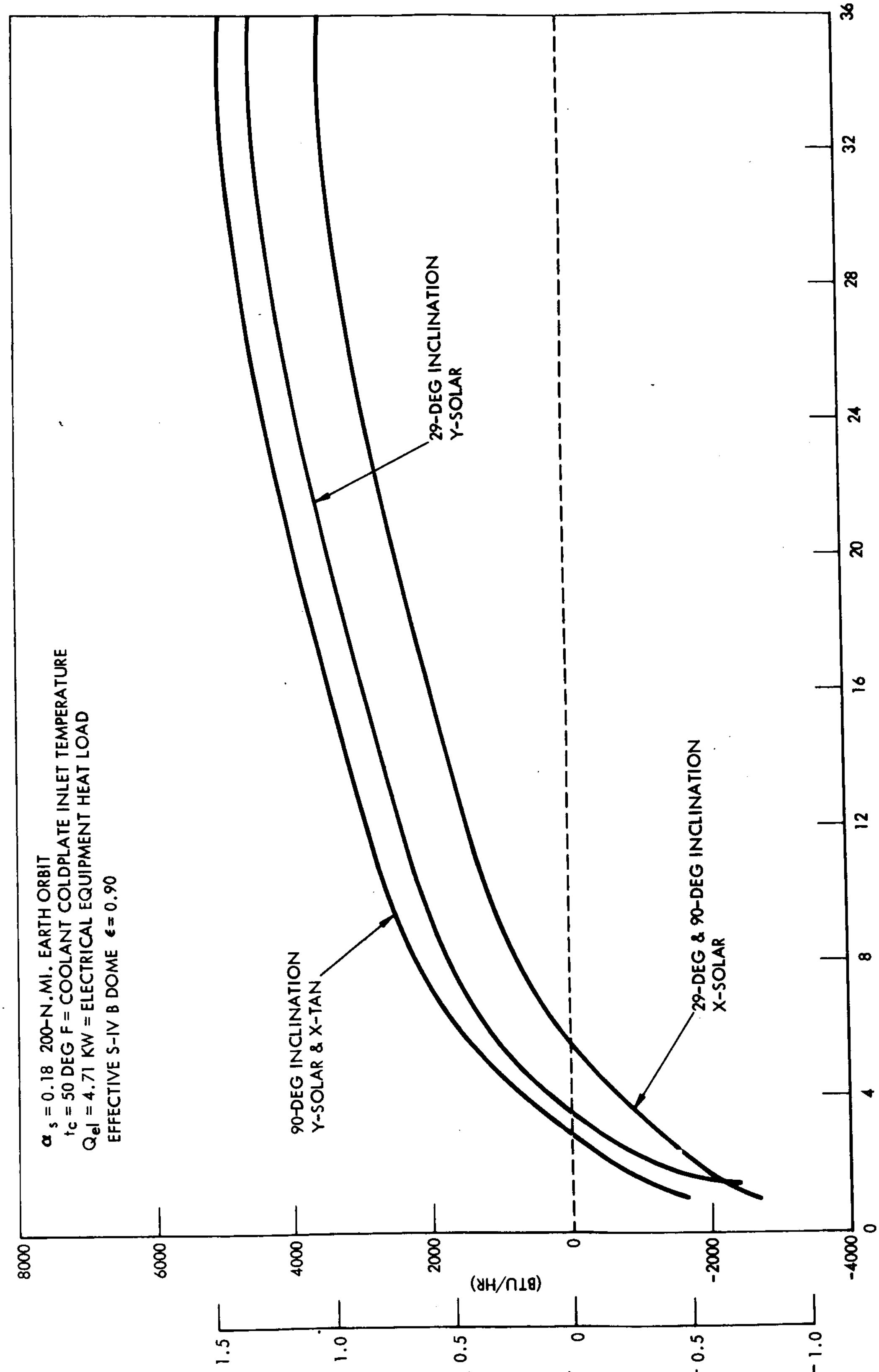

Figure A-13. Instrument Unit Net Heat Gain or Loss Versus Orbit Time, 200-N. Mi. Orbit, t_c = 50 F, Q_{el} = 4.71 kw, α_s = 0.18

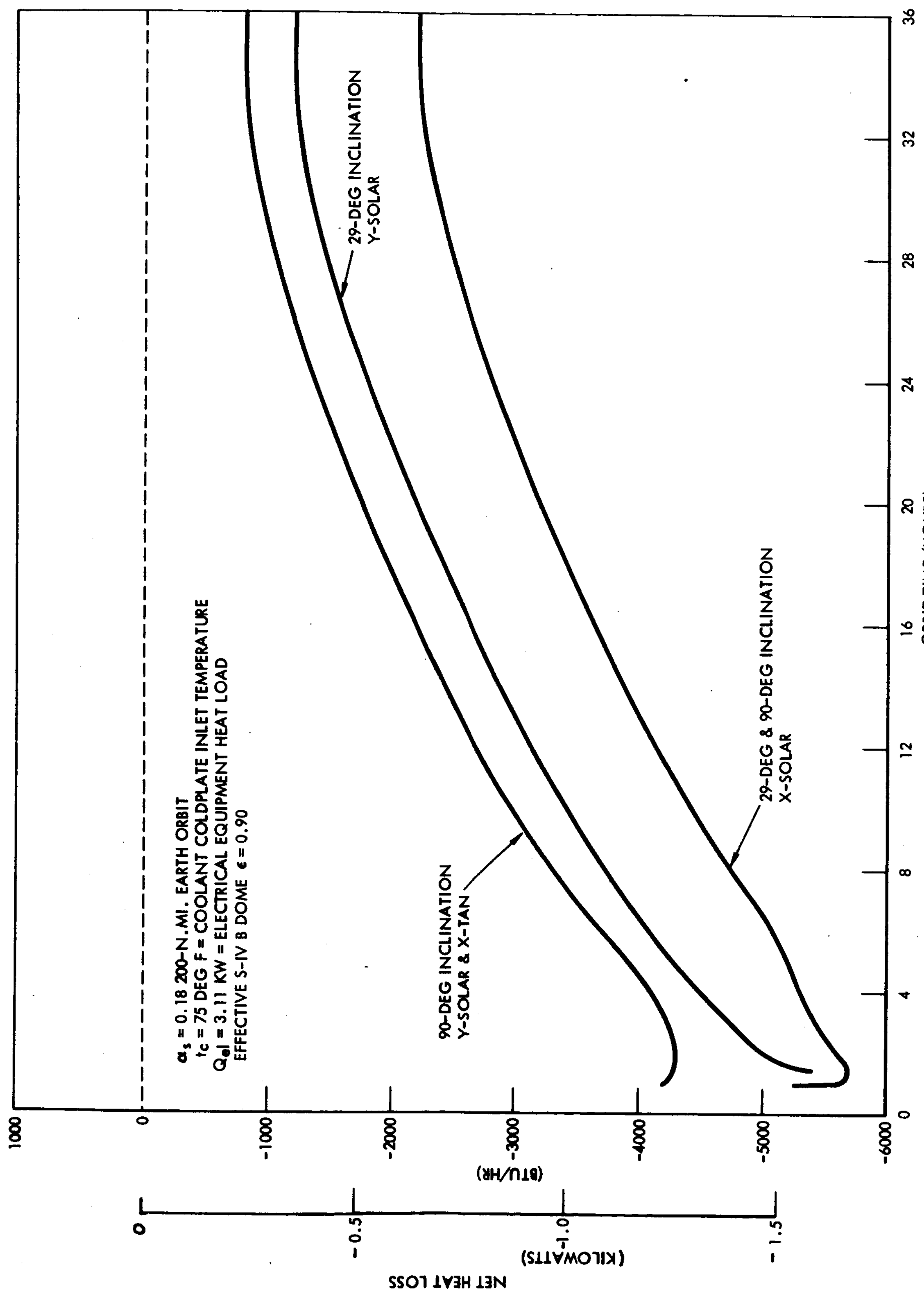

Figure A-14. Instrument Unit Net Heat Loss Versus Orbit Time, 200-N. Mi. Orbit, t_c = 75 F, Q_{el} = 3.11 kw, α_s = 0.18

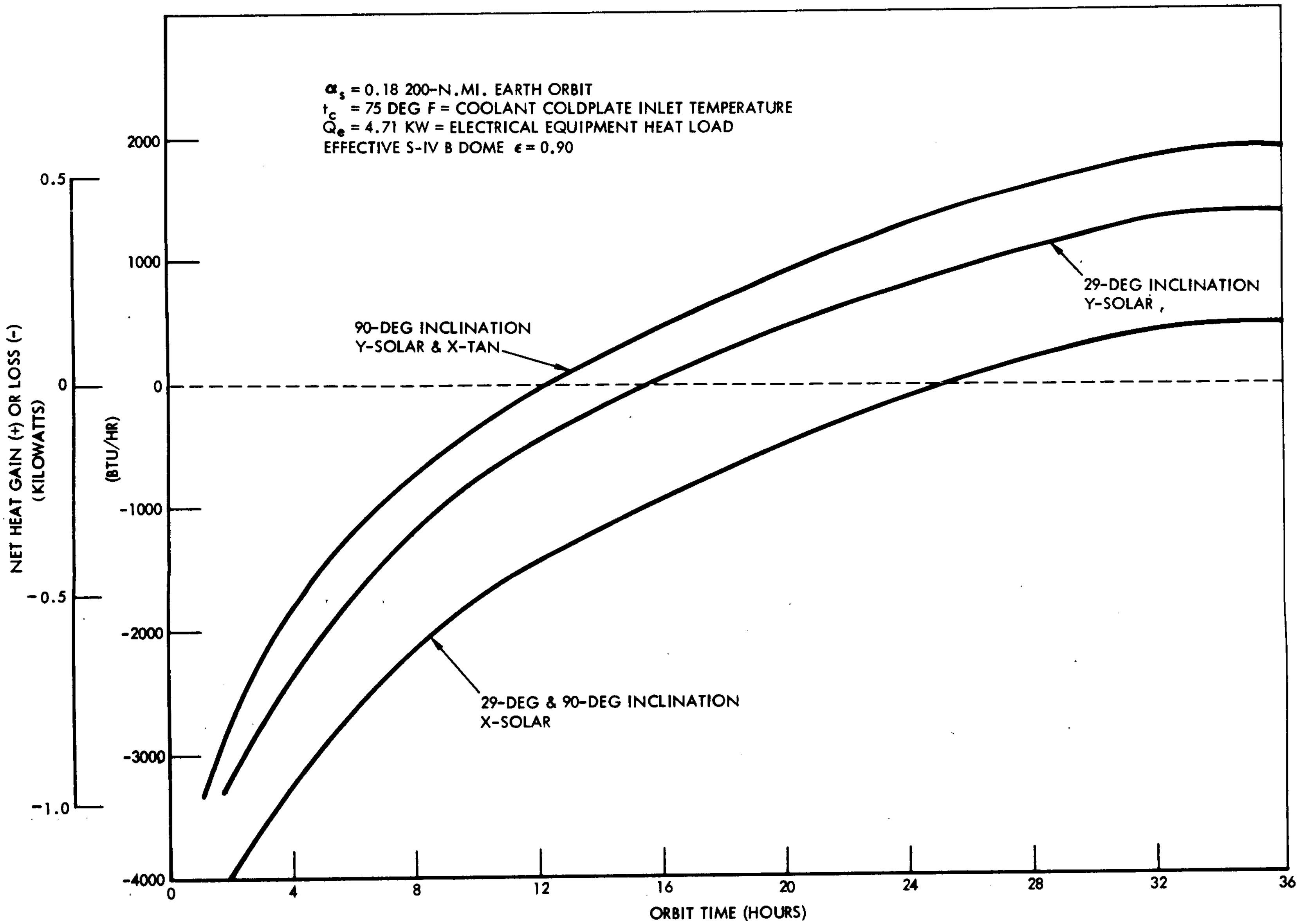

Figure A-15. Instrument Unit Net Heat Gain or Loss Versus Orbit Time, 200-N. Mi. Orbit, t_c = 75 F, Q_{el} = 4.71 kw, α_s = 0.18

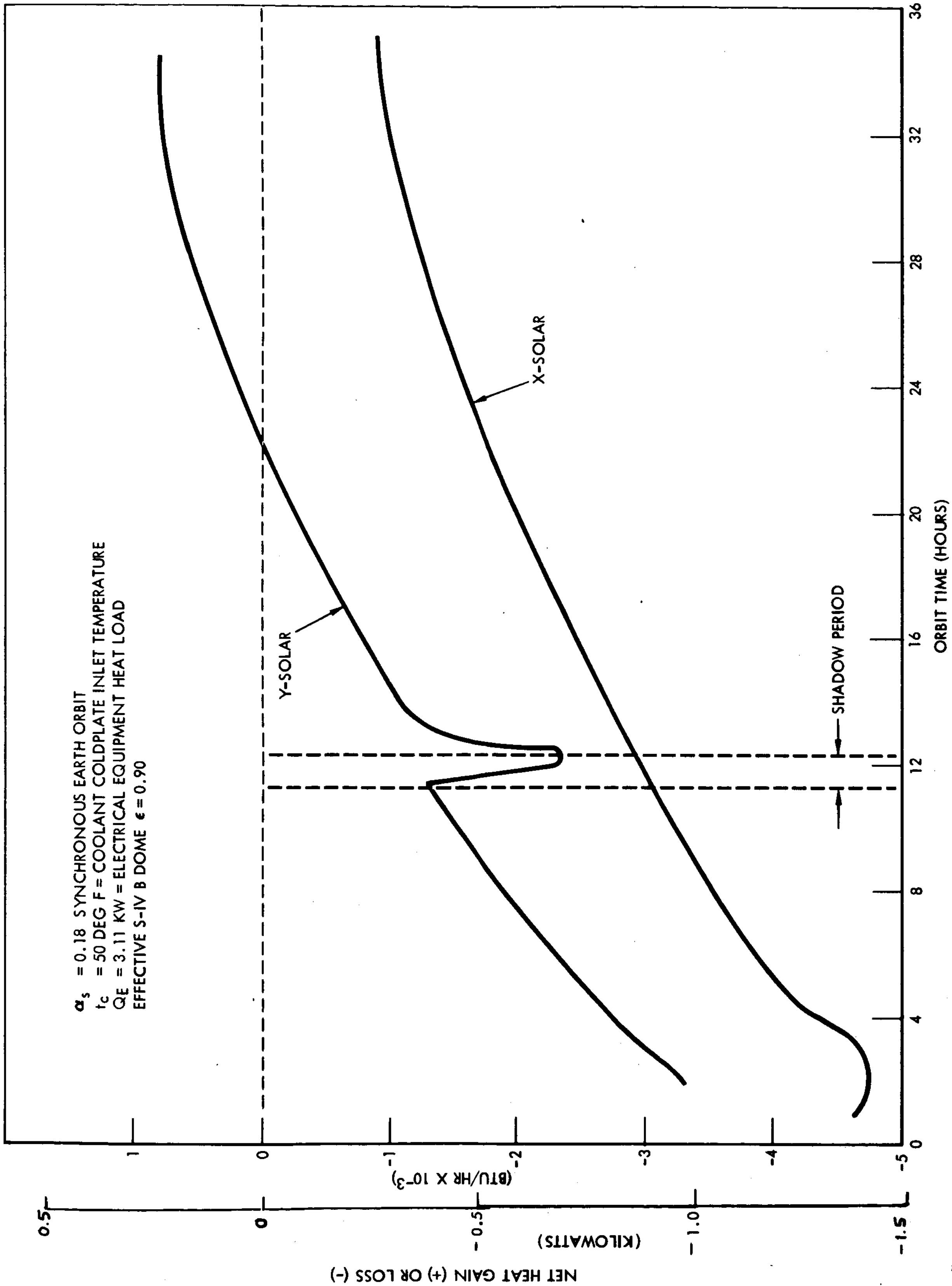

Figure A-16. Instrument Unit Net Heat Gain or Loss Versus Orbit Time, Synchronous Orbit, t_c = 50 F, Q_{el} = 3.11 kw, α_s = 0.18

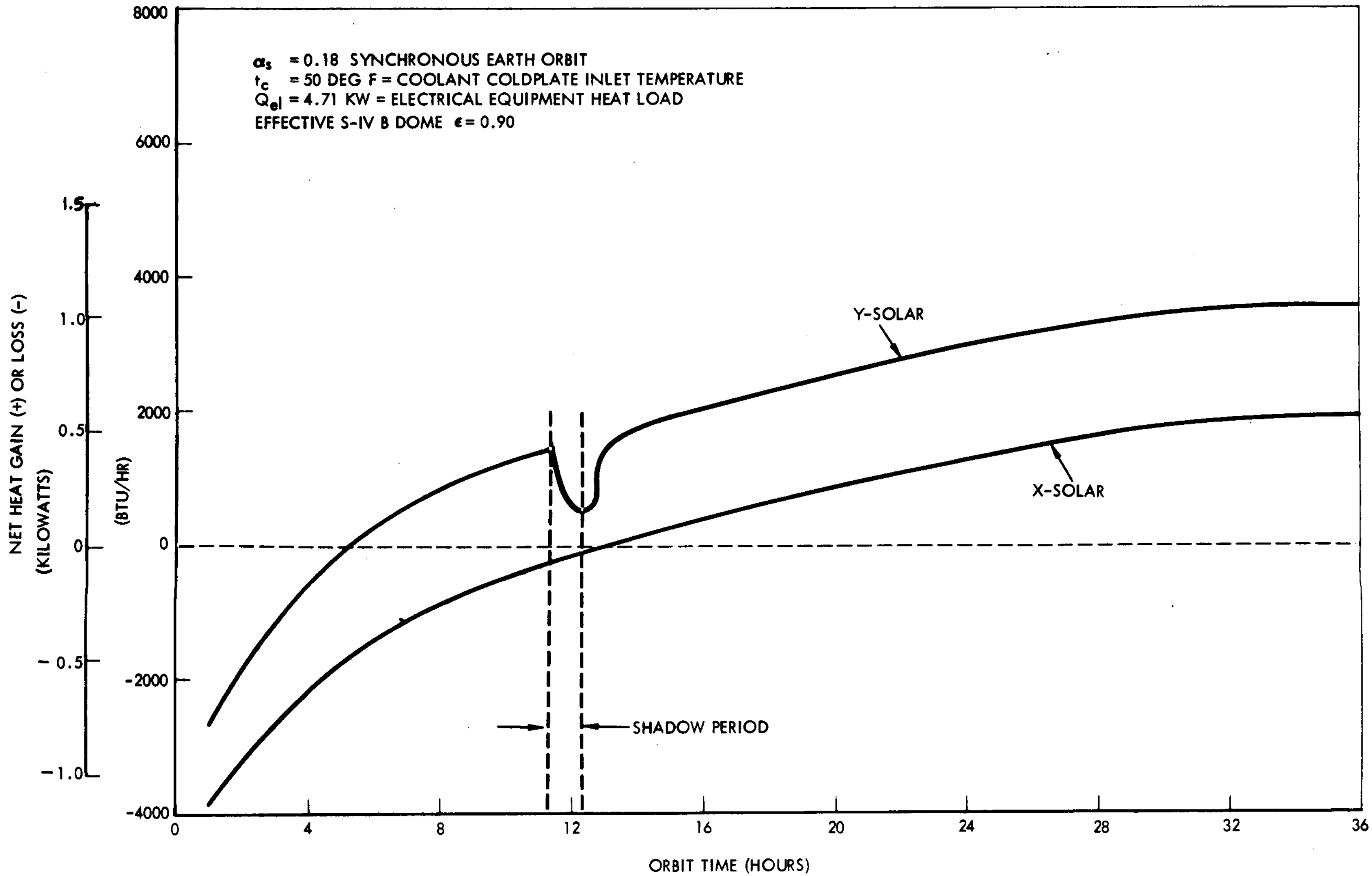

Figure A-17. Instrument Unit Net Heat Gain or Loss Versus Orbit Time, Synchronous Orbit, t_c = 50 F, Q_{el} = 4.71 kw, α_s = 0.18

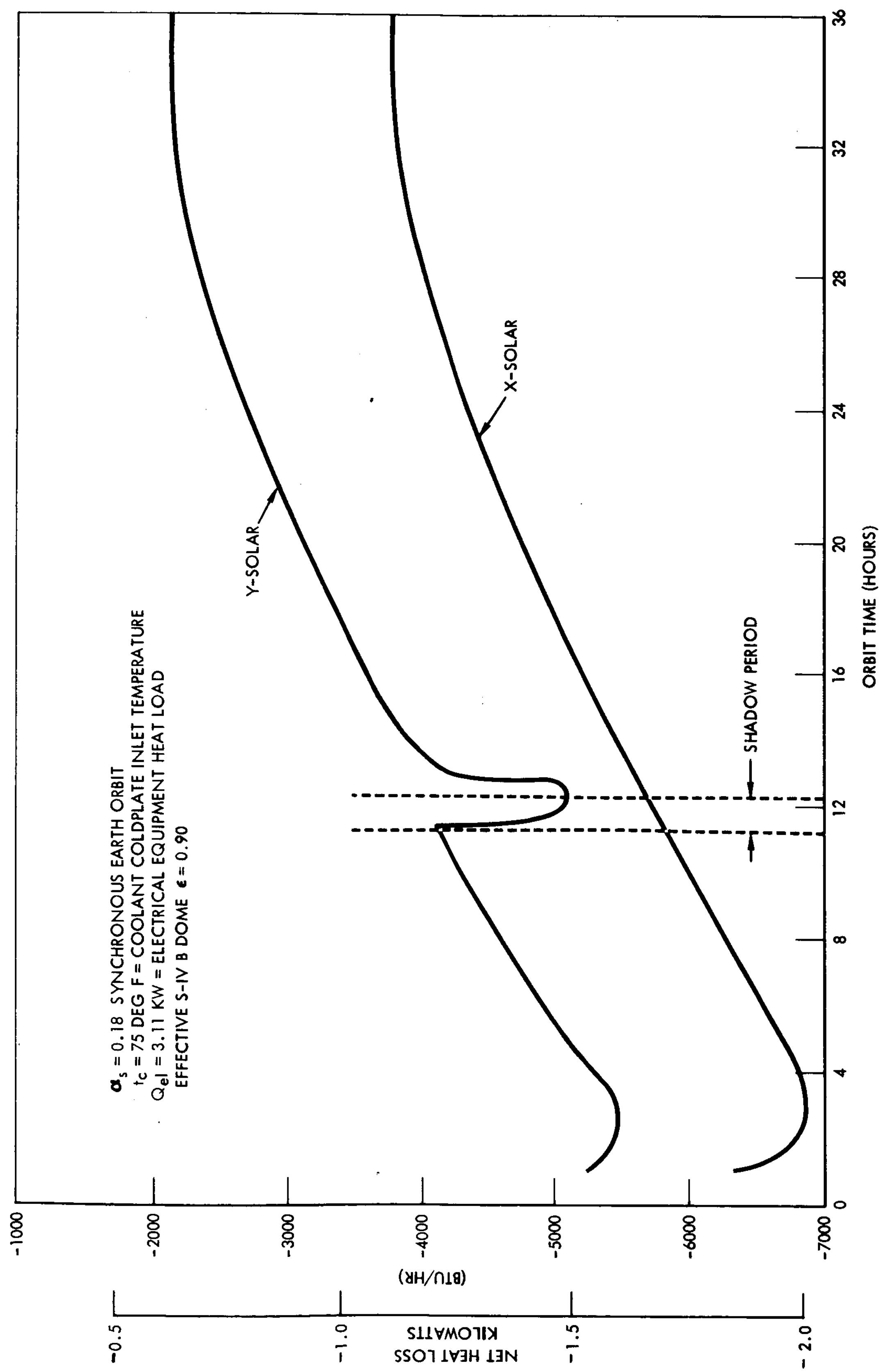

Figure A-18. Instrument Unit Net Heat Loss Versus Orbit Time, Synchronous Orbit, t_c = 75 F, Q_{el} = 3.11 kw, α_s = 0.18

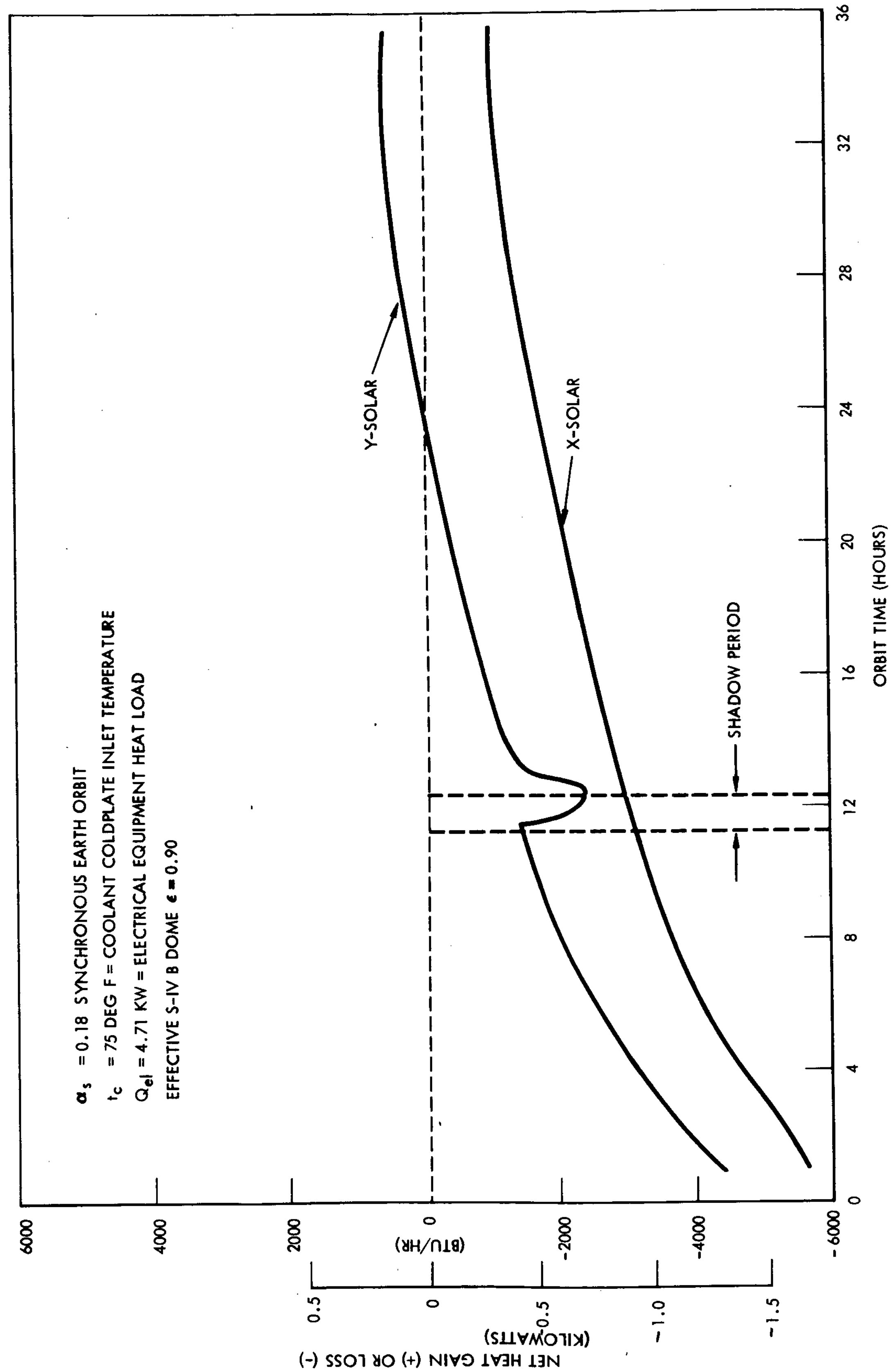

Figure A-19. Instrument Unit Net Heat Gain or Loss Versus Orbit Time, Synchronous Orbit, t_c = 75 F, Q_{el} = 4.71 kw, α_s = 0.18

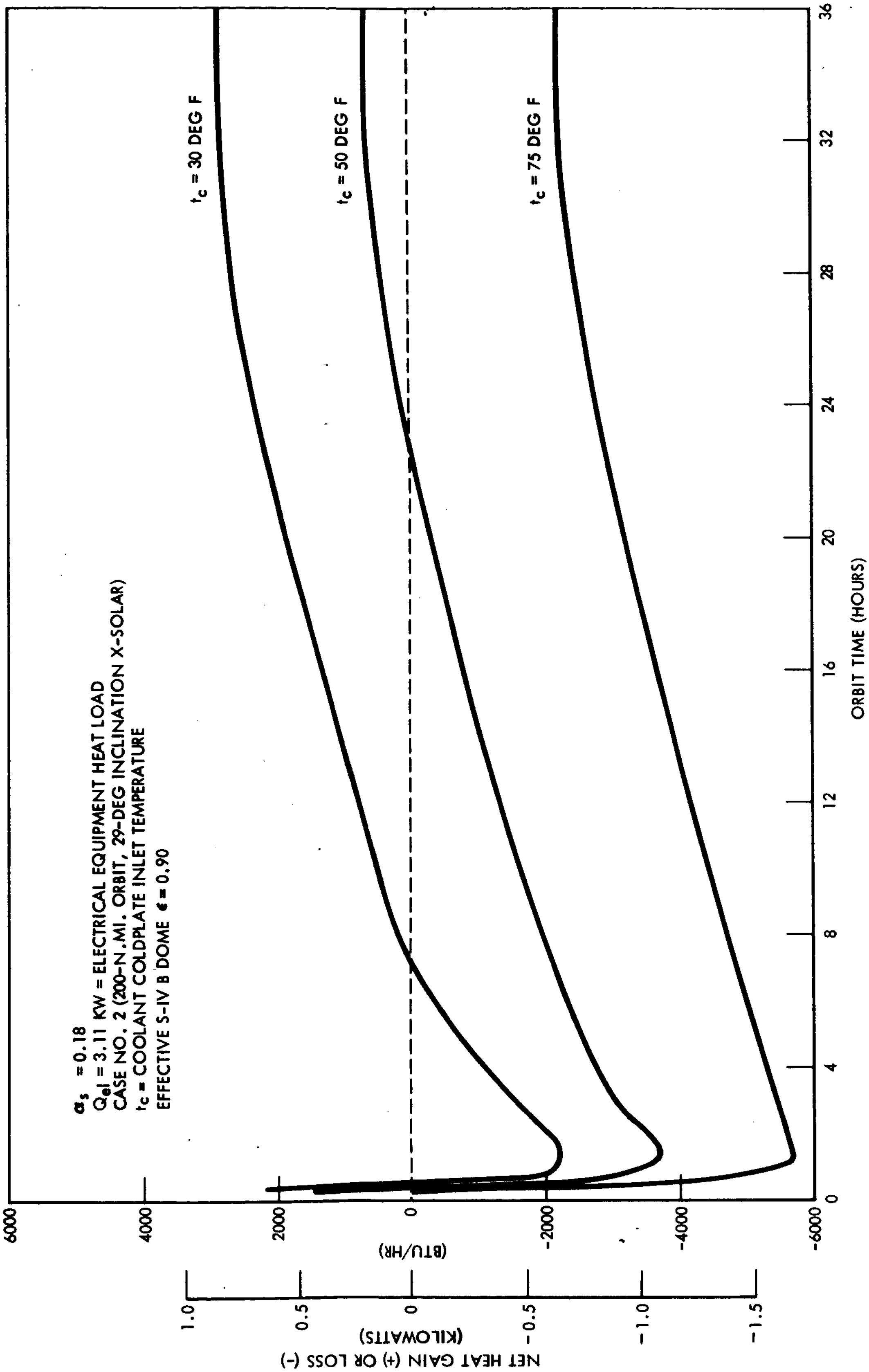

Figure A-20. Instrument Unit Net Heat Gain or Loss Versus Orbit Time, Case No. 2, Q_{el} = 3.11 kw

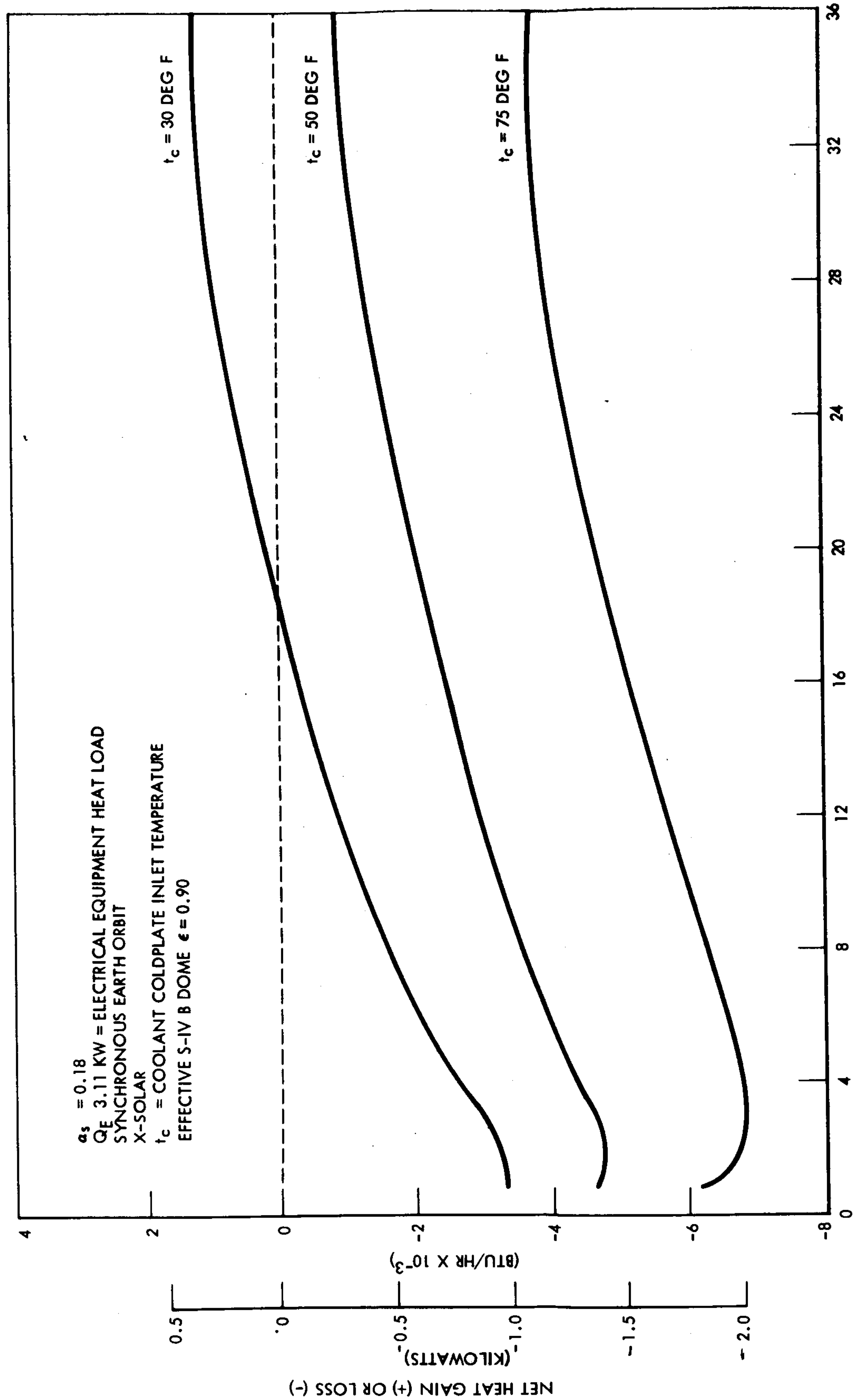

Figure A-21. Instrument Unit Net Heat Gain or Loss Versus Orbit Time, Synchronous Orbit, Q_{el} = 3.11 kw, X-Solar

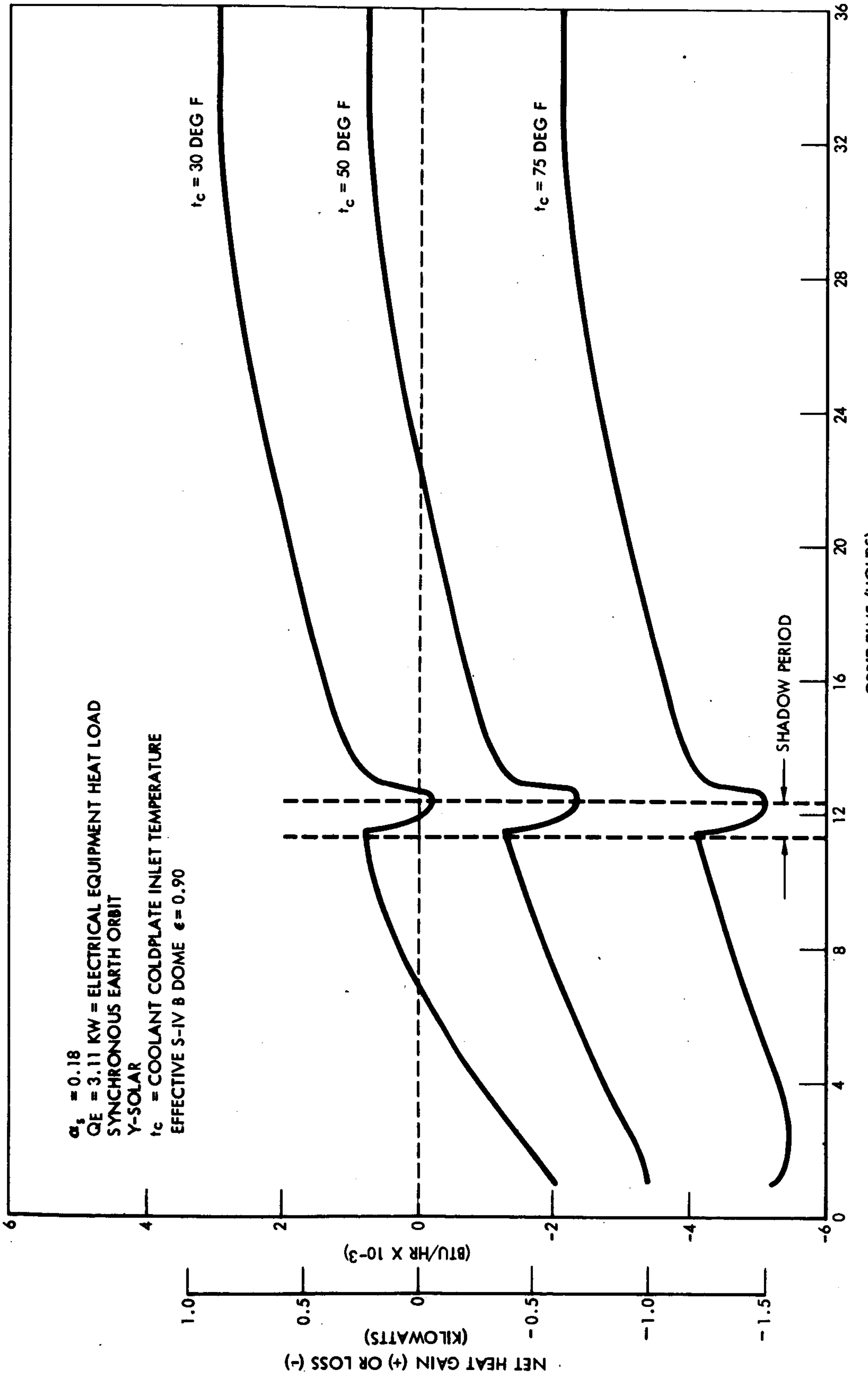

Figure A-22. Instrument Unit Net Heat Gain or Loss Versus Orbit Time, Synchronous Orbit, Q_{el} = 3.11 kw, Y-Solar

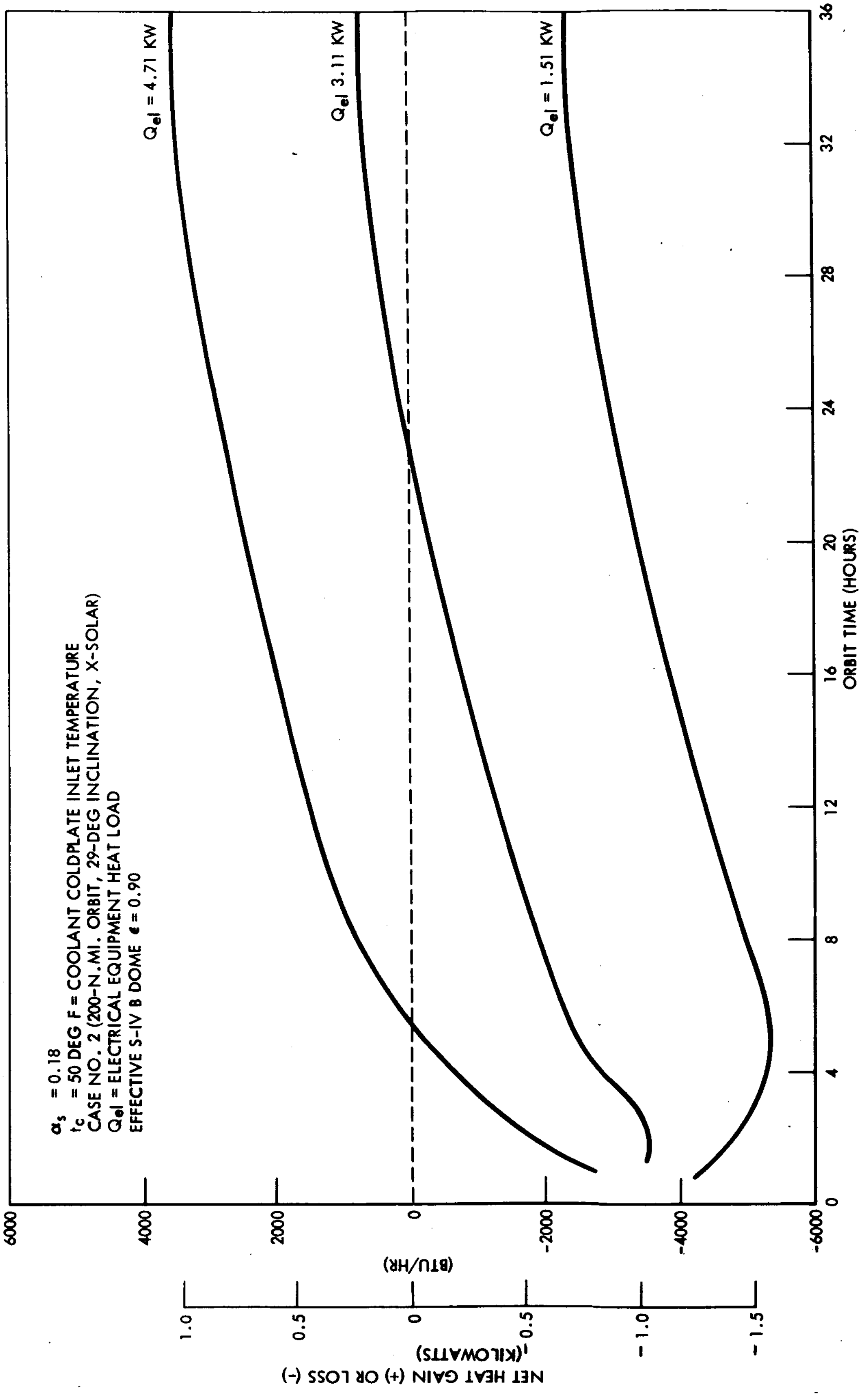

Figure A-23. Instrument Unit Net Heat Gain or Loss Versus Orbit Time, Case No. 2, t_c = 50 F

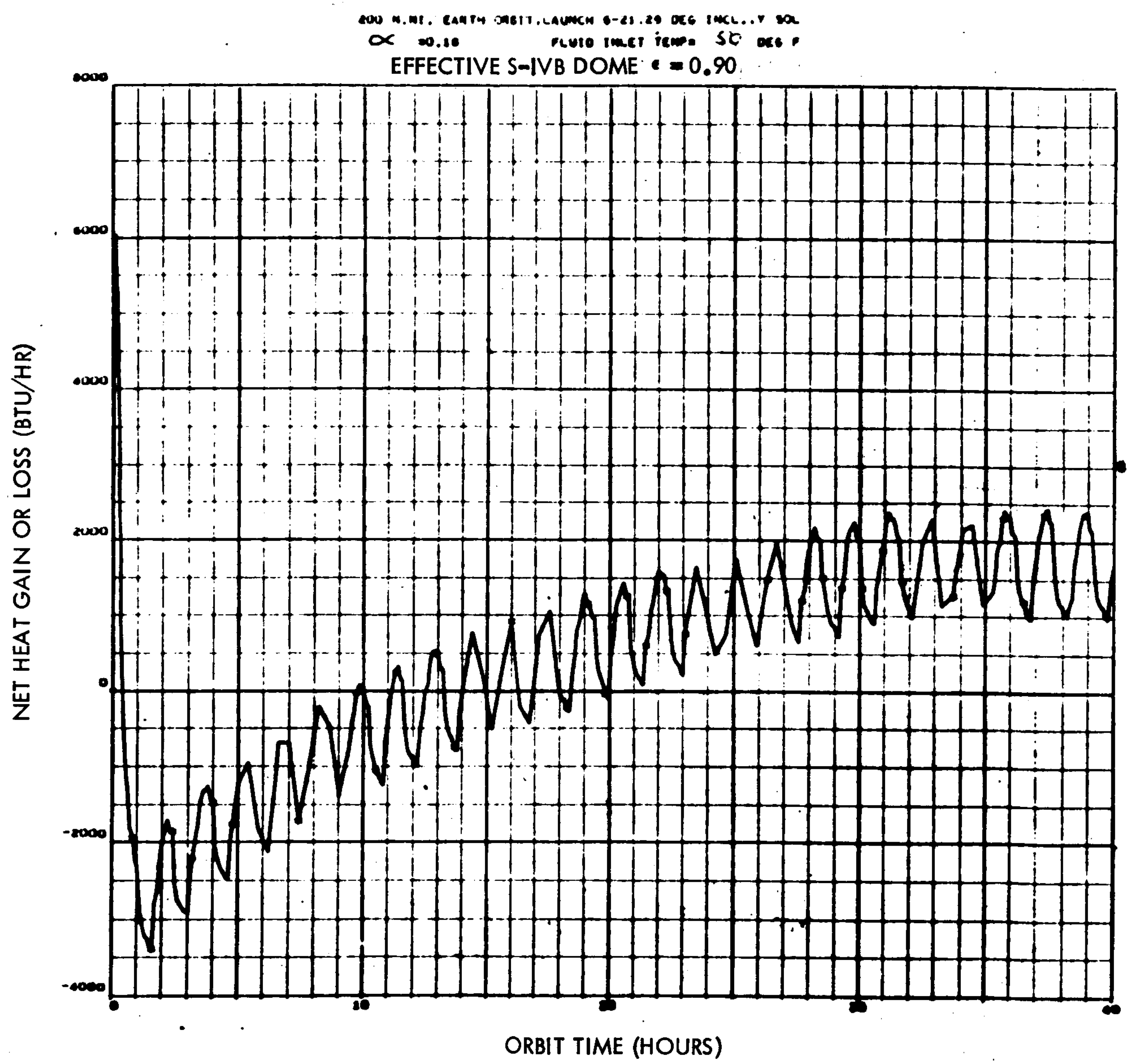

Figure A-24. Instrument Unit Net Heat Load, Solar Absorptivity = 0.18

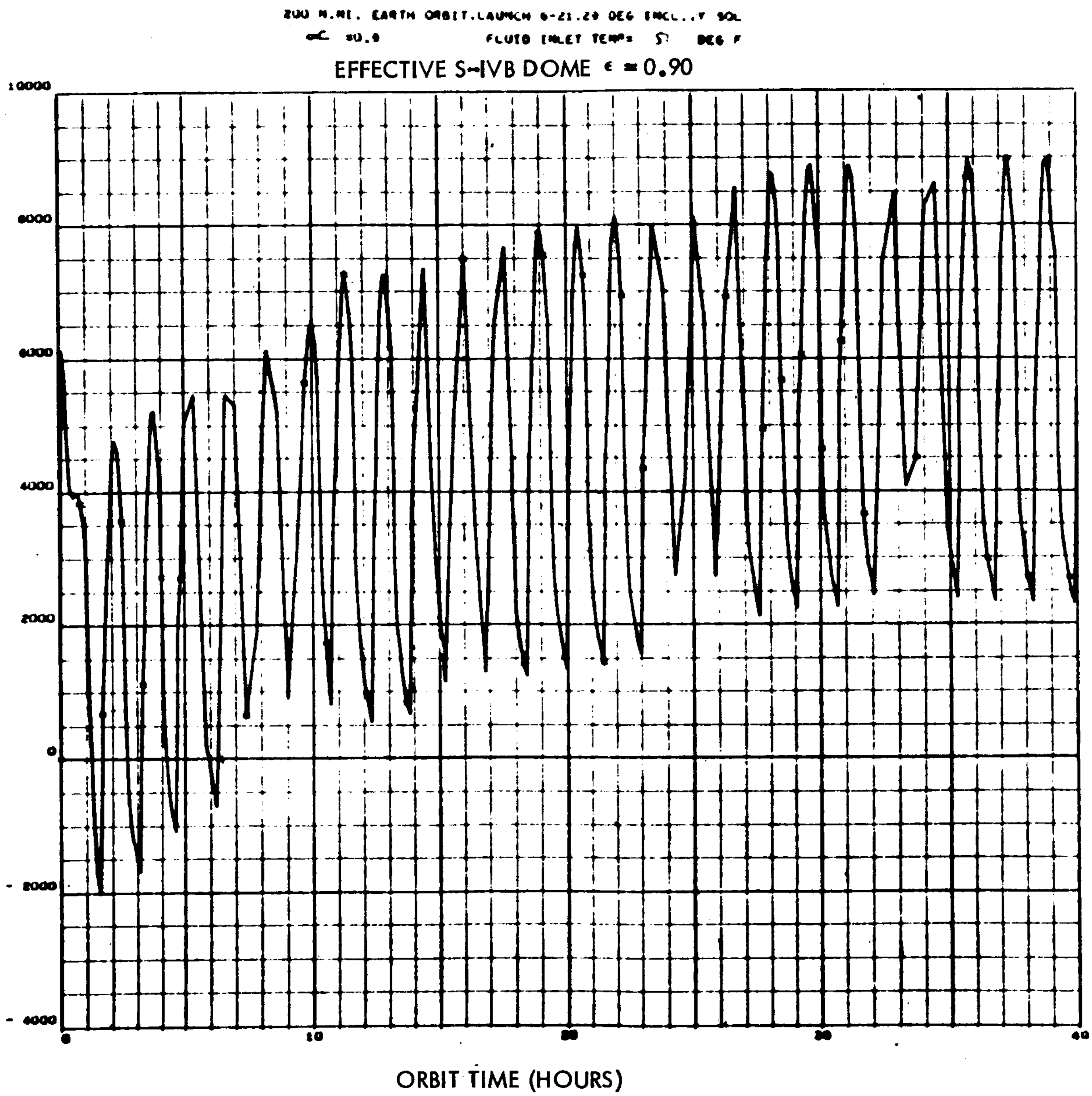

Figure A-25. Instrument Unit Net Heat Load, Solar Absorptivity = 0.9

Table A-1. Orbital Parameters

Case No.	Altitude (nautical miles)	Angle of Inclination	Launch Date	Spacecraft Orientation
IU-1a-1 IU-1b-2 IU-1c-3	200	29	June 21	X axis along orbit path X axis toward sun Y axis toward sun
IU-3a-4 IU-3b-5 IU-3c-6	200	90	March 21	X axis along orbit path X axis toward sun Y axis toward sun
IU-4a-7	100	0	March 21	X axis along orbit path
IU-6a-8 IU-6b-9 IU-6c-10	19,327 (synchronous)	0	March 21	Y axis toward sun X axis toward sun X axis along orbit path

circuit heat flows applicable to each coldplate and each integrally cooled piece of equipment.

The variation in IU heat load as a function of equipment power dissipation is shown in Figures A-4, A-5, and A-6 for constant coolant coldplate inlet temperature conditions of 30, 50, and 75 F, respectively. These plots also illustrate the effect of variation in α_s on average IU heatload. However, data for Cases 5 and 9, which represent solar orientation, indicate that the heat load for these cases is not affected by variation in α_s. The lines representing Case 2 in Figures A-4, A-5, and A-6 also represent results for Cases 5 and 8 when $\alpha_s = 0.18$, and the lines applicable to Case 3 at $\alpha_s = 0.18$ also represent Case 2 at $\alpha_s = 0.9$. Finally, data for Cases 3 and 4 also represent results for Cases 7 and 6, respectively.

Another method of presenting the variation in IU heat load as a function of IU outer shell solar absorptivity is illustrated in Figure A-7 for Case 3. These data are for a constant electrical equipment heat load of 3.11 kilowatts (150 watts per coldplate) and for coolant coldplate inlet temperatures of 30, 50, and 75 F. The variation in heat load with the variation α_s is practically linear. Hence, straight-line interpolation can be used for calculating IU heat loads at values of α_s between 0.18 and 0.9 from the data presented in Figures A-4, A-5, and A-6. Cross-plotting of the data in Figures A-4, A-5, and A-6 results in graphs showing IU heat load as a function of coolant coldplate inlet temperature, with electrical equipment heat load as a parameter. This type of plot is illustrated in Figures A-8 through A-11, which can be used to determine the coolant temperature range resulting from equipment heat load variation. For example, Figure A-8 shows that, when there is no heat rejection from or addition to the coolant circuit (zero IU heat load), the coolant temperature ranges from 28.5 F at Q_{el} = 1.5 kilowatts to 77 F at Q_{el} = 4.5 kilowatts. If this represents the possible variation in electrical equipment heat load for the orbital case in question and if the resulting coolant temperature range is permissible as far as equipment operation is concerned, no cooling or heating is necessary. Conversely, if the permissible coolant temperature range is defined, maximum cooling and/or heating requirements can be determined for a given equipment heat load variation. Thus, if the permissible coolant temperature range is assumed to be 40 F to 55 F and the equipment heat load varies between 1.5 and 3.5 kilowatts, Figure A-8 indicates a maximum heating requirement of 1250 Btu's per hour and a maximum cooling requirement of 700 Btu's per hour.

The data discussed to this point refer to equilibrium environmental conditions—i.e., when the S-IVB dome temperature has reached its final value of 60 F. To illustrate conditions applicable to operation between injection into orbit and the time when equilibrium is reached (0 to 35 hours of orbit time), Figures A-2 through A-19 summarize the transient conditions of 200-nautical-mile and synchronous earth orbits. These figures show IU heat load as a function of orbit time for coolant coldplate inlet temperatures of 50 and 75 F and for electrical equipment heat loads of 3.11 and 4.71 kilowatts. For all operating conditions selected in this investigation, heat must be added to the coolant circuit during the initial portion of each mission. This requirement is apparent from an examination of Figures A-12 through A-19, which also indicate that some missions will require continuous heat addition. Data for the synchronous orbit with the X axis along the orbit path (x-tan) have not been plotted in Figures A-17, A-18, and A-19. The heating or cooling requirements for this orbital condition will generally fall between those for the two other conditions shown (x-solar and y-solar), as indicated in Figure A-16. It should also be noted that the dip in some of the curves, beginning at an orbit time of 11.4 hours, corresponds to the passage of the IU through the earth's shadow.

Figures A-20, A-21, and A-22 illustrate another method of presenting data for the transient condition. They show IU heat load versus orbit time, with coolant coldplate inlet temperature as a parameter. These graphs apply to a constant electrical equipment heat load of 3.11 kilowatts and to three specific orbit conditions. For the 200-nautical-mile orbit, Figure A-23 shows IU heat load versus orbit time, with electrical equipment heat load as a parameter. This graph is based on a constant coolant coldplate inlet temperature of 50 F.

Data similar to those shown in Figures A-12 through A-19 are presented in Figures A-26 through A-33. The latter results were obtained under the same conditions of orbital altitude, vehicle orientation, electrical equipment heat loads, and coolant coldplate inlet temperature as the former data, but a solar absorptivity value of 0.9 was used for the instrument unit outer shell instead of the previous value of 0.18. Bearing in mind that the net heat gain or loss shown in the graphs represents the average during a complete orbit, a comparison can be made to evaluate the effect of change in solar absorptivity on instrument unit heat load during varying S-IVB dome temperature conditions. This comparison reveals the following:

1. The shape of the curves of net heat gain or loss as a function of time is not affected by variation in IU outer shell solar absorptivity.

2. The instrument unit heat load is not affected by variation in IU outer shell solar absorptivity when the IU is in the minimum solar heating orientation (x-solar) and in a synchronous orbit or a 200-nautical-mile polar orbit. This is due to the absence of any direct solar radiation incident upon the IU outer shell and the negligible effect of albedo radiation.

3. When the IU is in the minimum solar heating orientation (x-solar) and in a 200-nautical-mile earth orbit at 90-degrees inclination, a change in IU outer shell solar absorptivity from 0.18 to 0.9 results in an increase in instrument unit heat load of 1000 Btu/hr. This increase is due to albedo radiation alone because, in the solar orientation, there is no direct solar radiation incident upon the IU outer shell.

4. When the IU is in the maximum solar heating orientation (y-solar) and in a 200-nautical-mile earth orbit at 29-degrees inclination, a change in IU outer shell solar absorptivity from 0.18 to 0.9 results in an increase in instrument unit heat load of 4000 Btu/hr for a coolant coldplate inlet temperature of 50 F. When this temperature is 75 F, the increase in heat load varies from 3700 to 3950 Btu/hr from the beginning to the end of the period investigated.

5. When the IU is in the maximum solar heating orientation (y-solar) and in a 200-nautical-mile polar orbit, a change in IU outer shell solar absorptivity from 0.18 to 0.9 results in an increase in instrument unit heat load of 5500 Btu/hr for a coolant coldplate inlet temperature of 50 F. When this temperature is 75 F, the increase in heat load varies from 4750 to 5400 Btu/hr, depending on orbit time and electrical equipment heat load level.

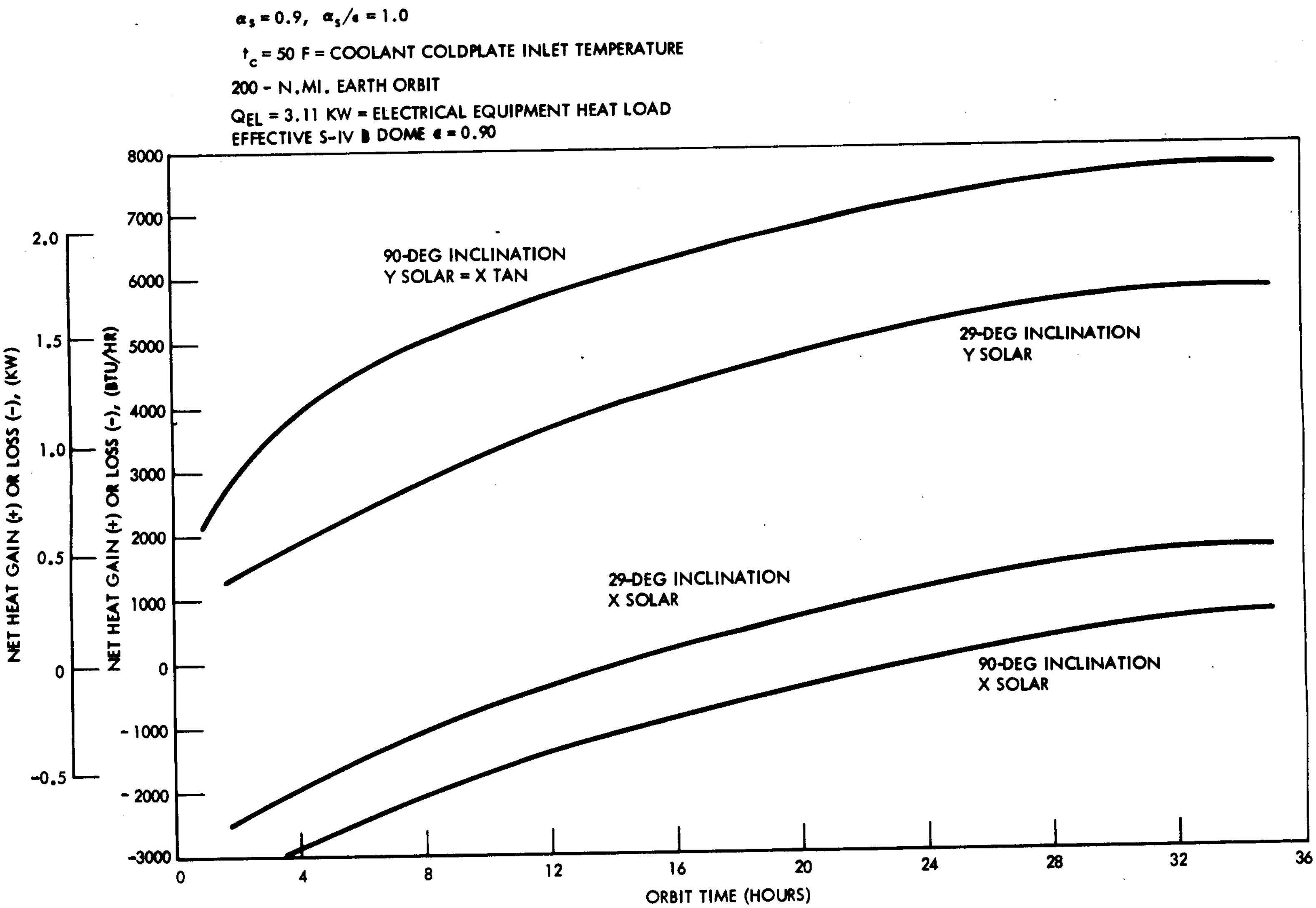

Figure A-26. Instrument Unit Net Heat Gain or Loss Versus Orbit Time, 200-N. Mi. Orbit, t_c - 50 F, Q_{el} = 3.11 kw, α_s = 0.9

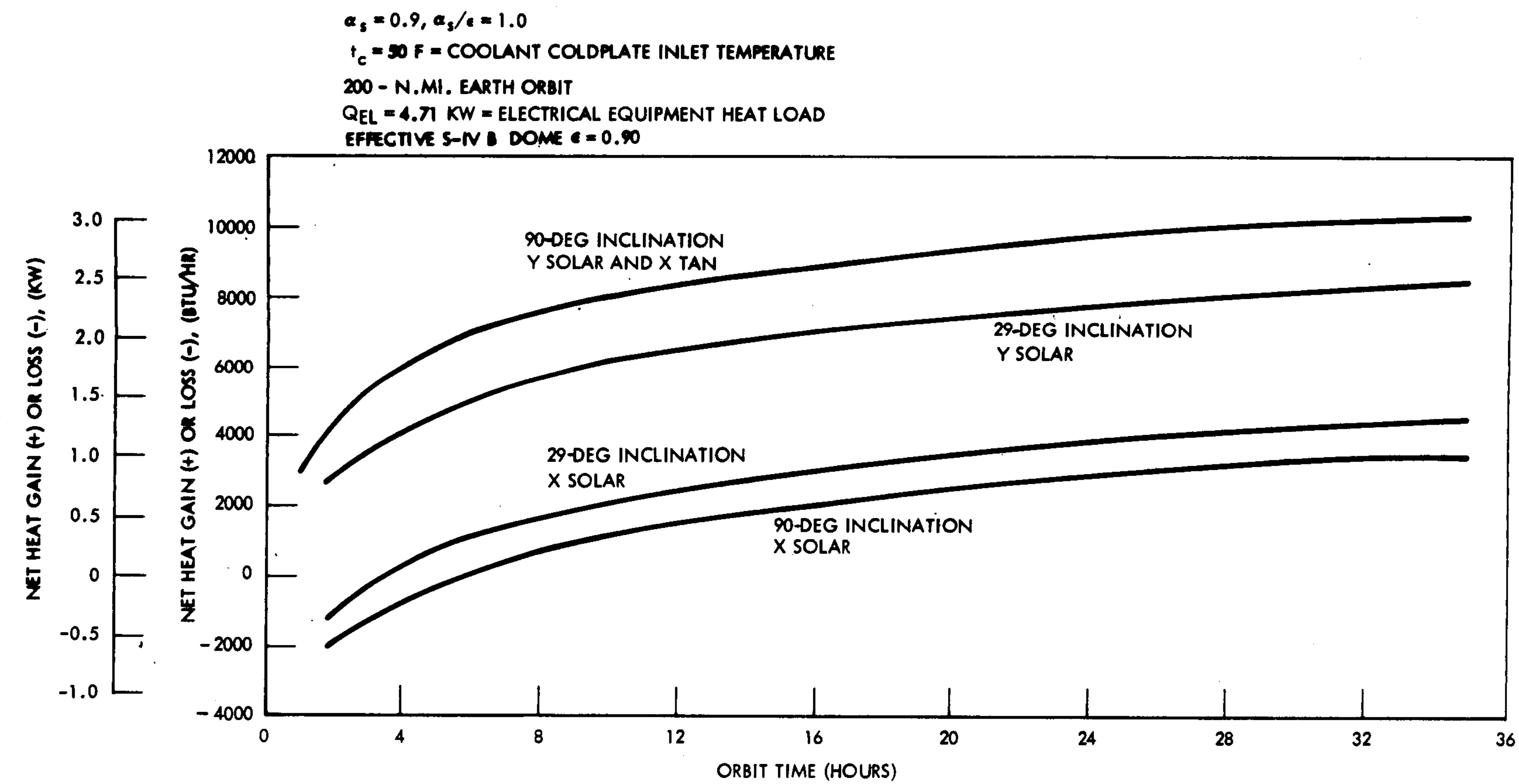

Figure A-27. Instrument Unit Net Heat Gain or Loss Versus Orbit Time, 200-N. Mi. Orbit, t_c = 50 F, Q_{el} = 4.71 kw, α_s = 0.9

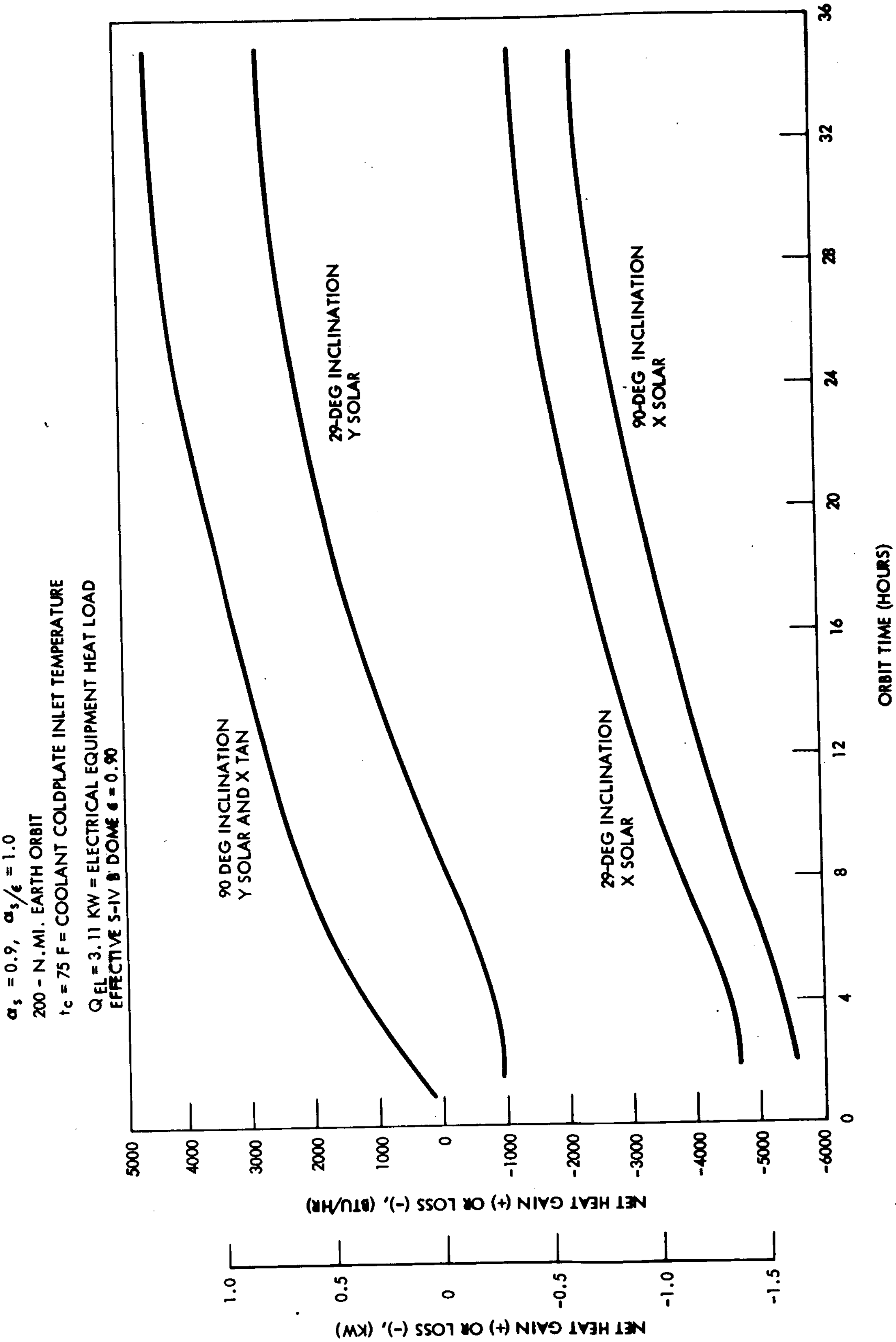

Figure A-28. Instrument Unit Net Heat Gain or Loss Versus Orbit Time, 200-N. Mi. Orbit, t_c = 75 F, Q_{el} = 3.11 kw, α_s = 0.9

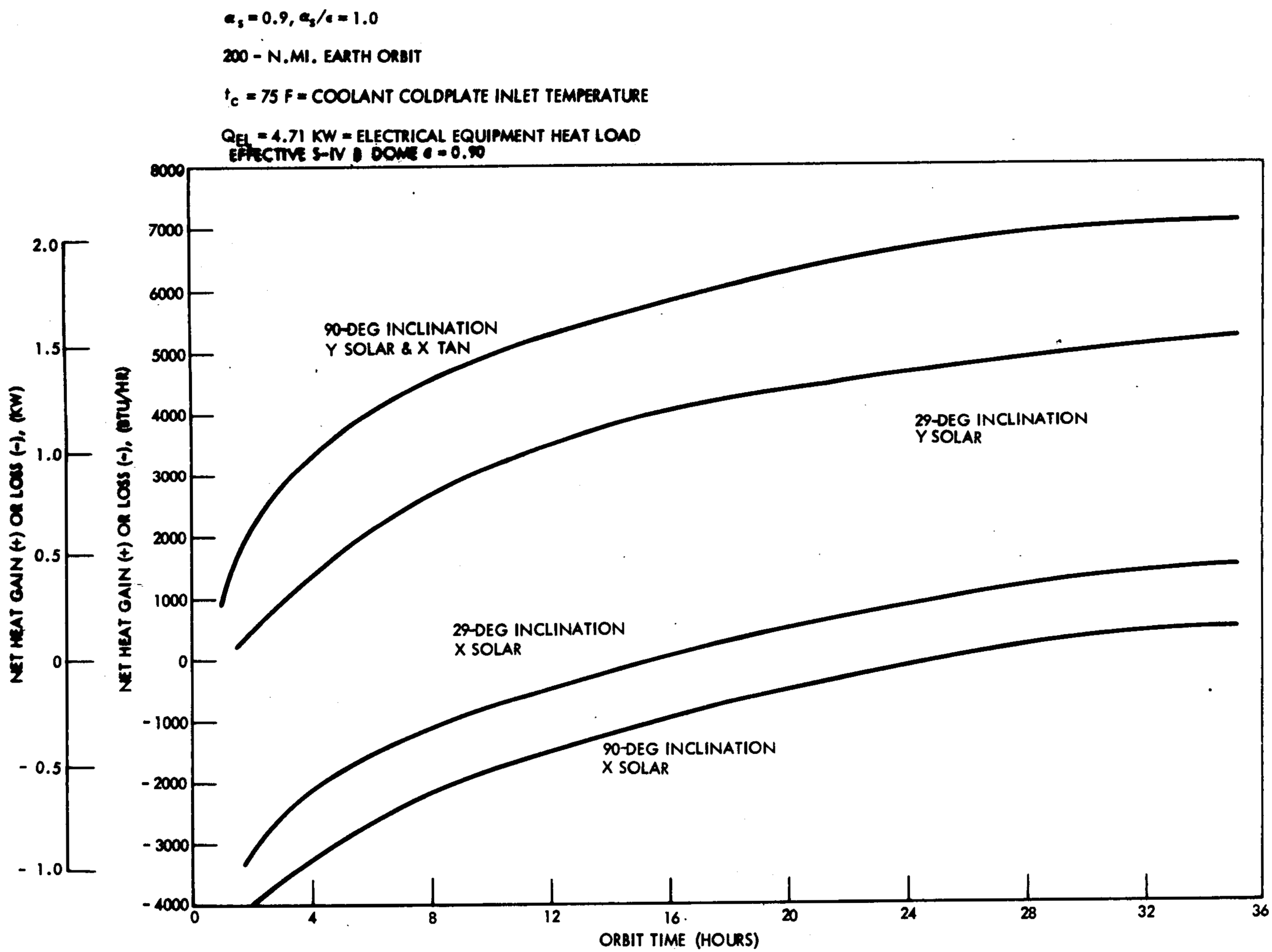

Figure A-29. Instrument Unit Net Heat Gain or Loss Versus Orbit Time, 200-N. Mi. Orbit, t_c = 75 F, Q_{el} = 4.71 kw, α_s = 0.9

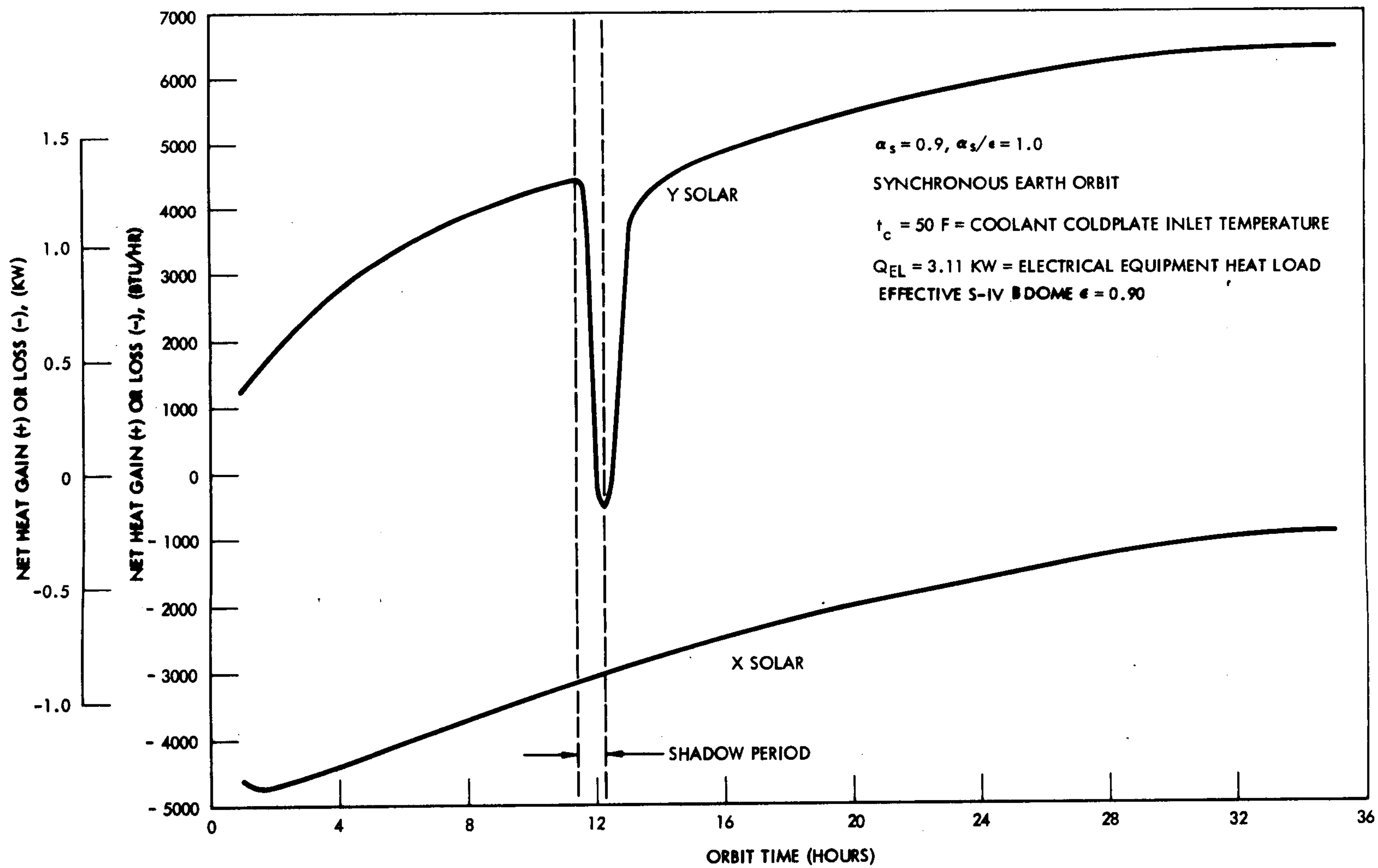

Figure A-30. Instrument Unit Net Heat Gain or Loss Versus Orbit Time, Synchronous Orbit, t_c = 50 F, Q_{el} = 3.11 kw, α_s = 0.9

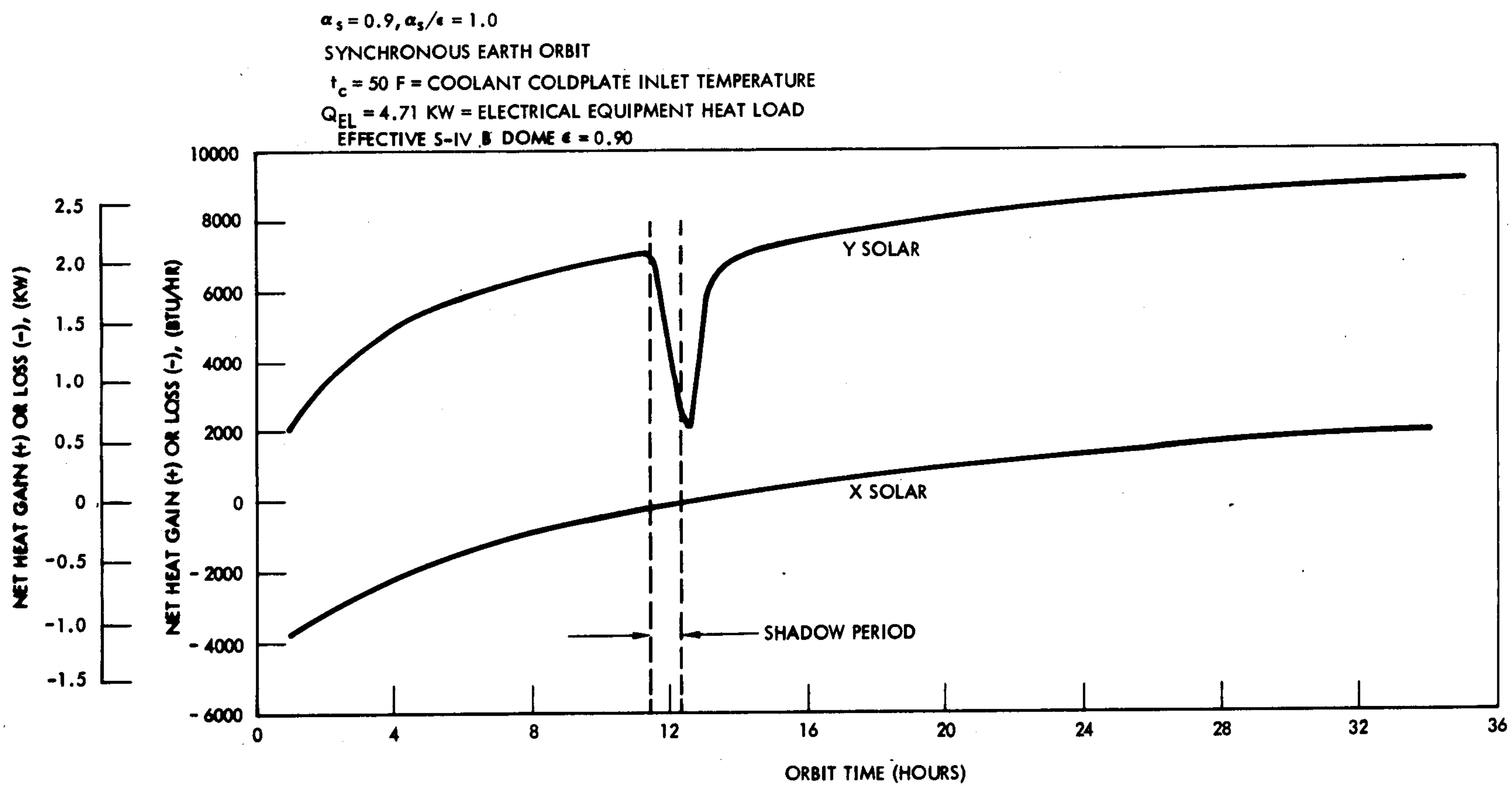

Figure A-31. Instrument Unit Net Heat Gain or Loss Versus Orbit Time, Synchronous Orbit, t_c = 50 F, Q_{el} = 4.71 kw, α_s = 0.9

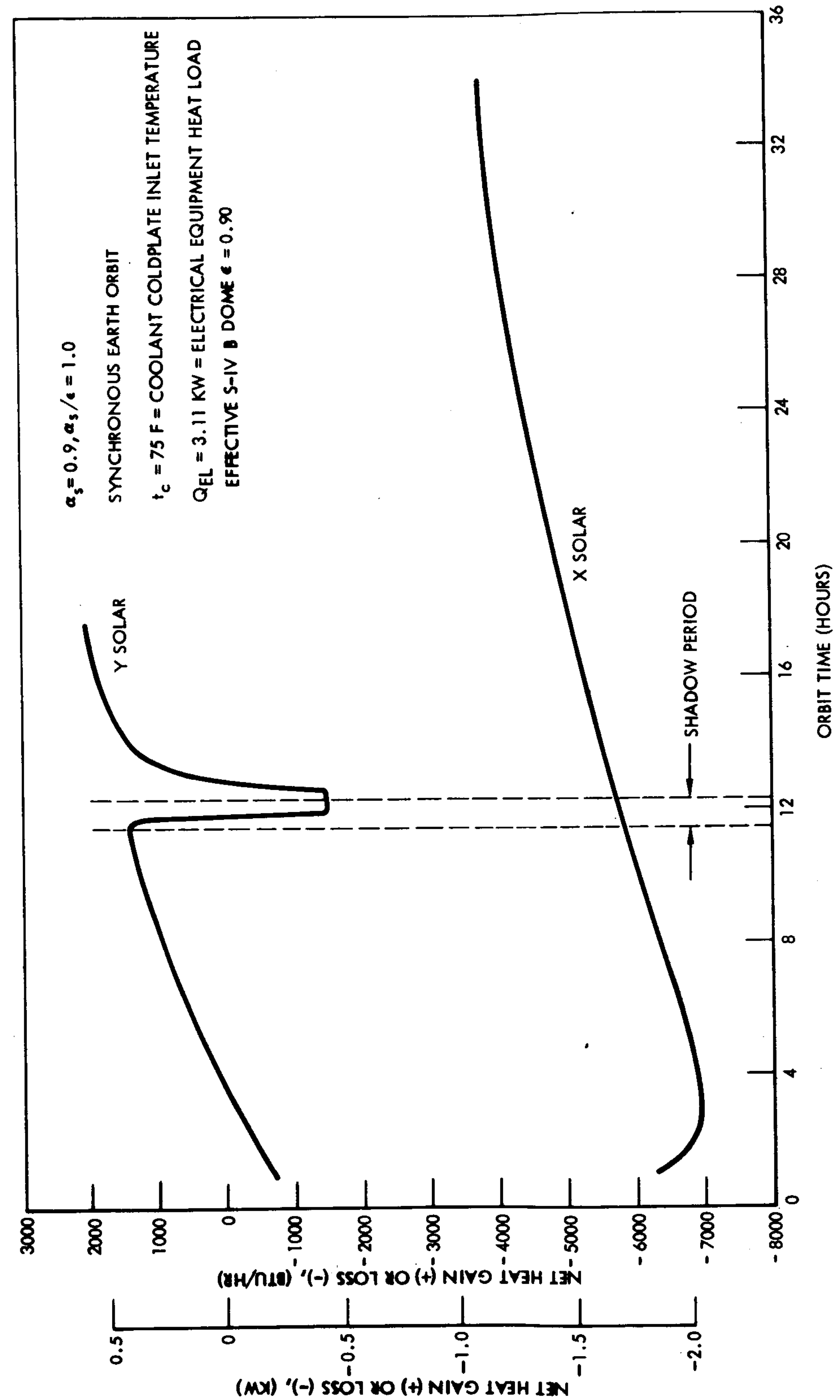

Figure A-32. Instrument Unit Net Heat Gain or Loss Versus Orbit Time, Synchronous Orbit, t_c = 75 F, Q_{el} = 3.11 kw, α_s = 0.9

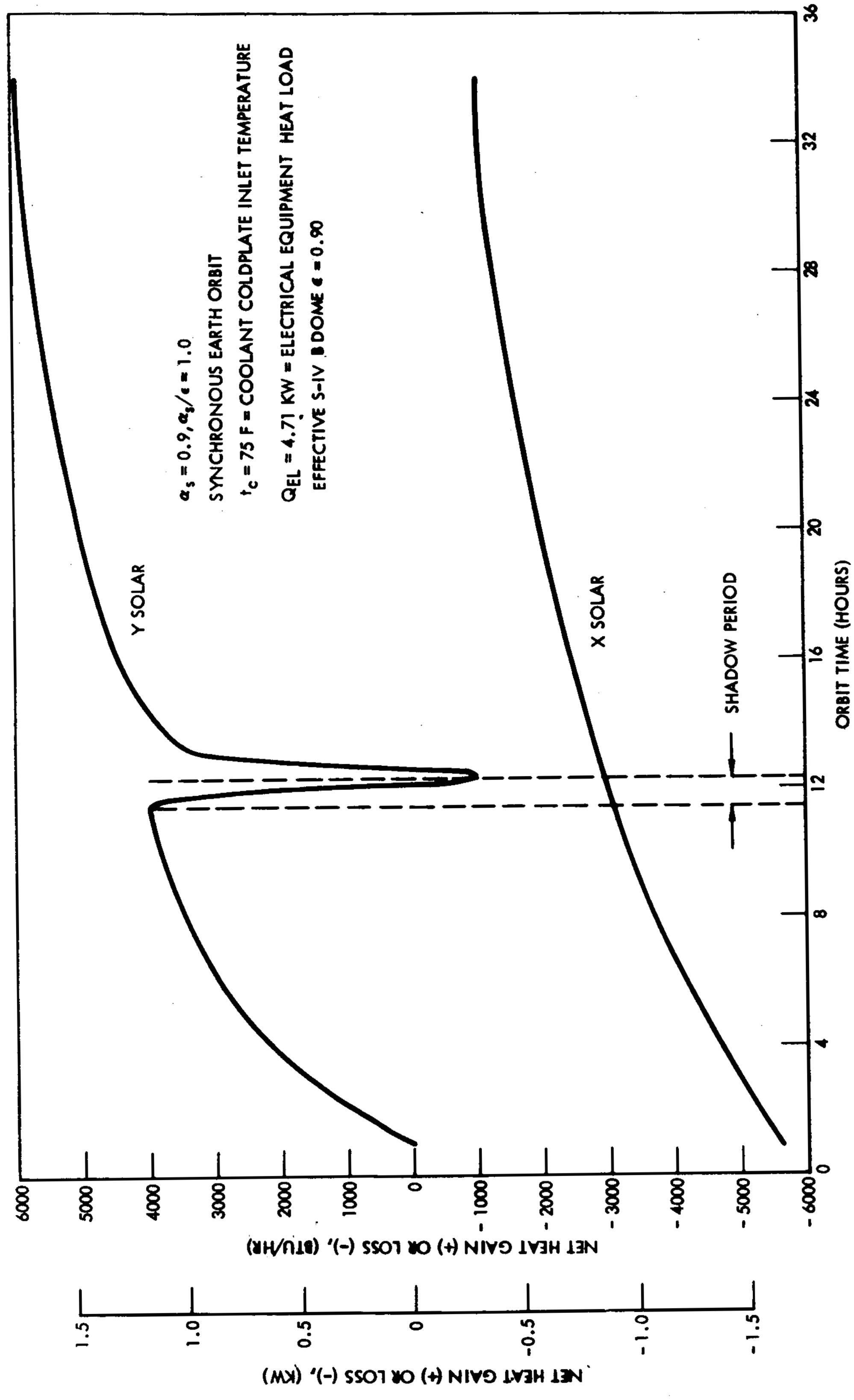

Figure A-33. Instrument Unit Net Heat Gain or Loss Versus Orbit Time, Synchronous Orbit, t_c = 75 F, Q_{el} = 4.71 kw, α_s = 0.9

6. When the IU is in the maximum solar heating orientation (y-solar) and in a synchronous orbit, a change in IU outer shell solar absorptivity from 0.18 to 0.9 results in an increase in instrument unit heat load of 5400-5700 Btu/hr, depending on coolant coldplate inlet temperature and electrical equipment heat load level.

These results, which have been summarized in Table A-2, suggest the possibility of using the solar absorptivity of the IU outer shell to counteract the influence of the S-IVB dome during the first few hours of a mission. During 200-nautical-mile orbits, however, higher values of solar absorptivity produce large variations in solar heating rate with change in orbital position. As a result, control system requirements may become difficult to meet if the coolant temperature needs to be maintained within a narrow band.

Table A-2. Instrument Unit Net Heat Increment Due to α_s Change from 0.18 to 0.9

Mission	Net Heat Increment (Btu/hr)			
	Coolant Coldplate Inlet Temp. (50 F)		Coolant Coldplate Inlet Temp. (75 F)	
	Electrical Equipment Heat Load (kw)		Electrical Equipment Heat Load (kw)	
	3.11	4.71	3.11	4.71
Synchronous orbit x-solar	No Change			
200-n. mi. polar orbit x-solar	No Change			
200-n. mi. orbit, 29 degrees inclination x-solar	+1000	+1000	+1000	+1000
200-n. mi. orbit, 29 degrees inclination y-solar	+4000	+4000	+3700 to +3950	+3700 to +3900
200-n. mi. polar orbit y-solar	+5500	+5500	+5400	+4750 to +5200
Synchronous orbit y-solar	+5700	+5500	+5500	+5400

For instrument unit orientations which are not affected by variations in solar absorptivity of the outer shell, and for those cases in which the increase in solar heating with increase in α_S is less than desired, the temperature level of the coolant may be raised by using an outer shell coating with an infrared emissivity lower than the value of 0.9 used in the thermal analysis. The effect of variation in emissivity on instrument unit total heat load is illustrated in Figure A-34, which represents data for the synchronous orbit condition and with the instrument unit in a minimum solar heating orientation (x-solar). The instrument unit heat loads were calculated for a constant equipment heat load of 3.11 kilowatts and coolant coldplate inlet temperatures of 30, 50, and 75 F. The strong influence of emissivity on net heat gain or loss is readily seen in the illustration.

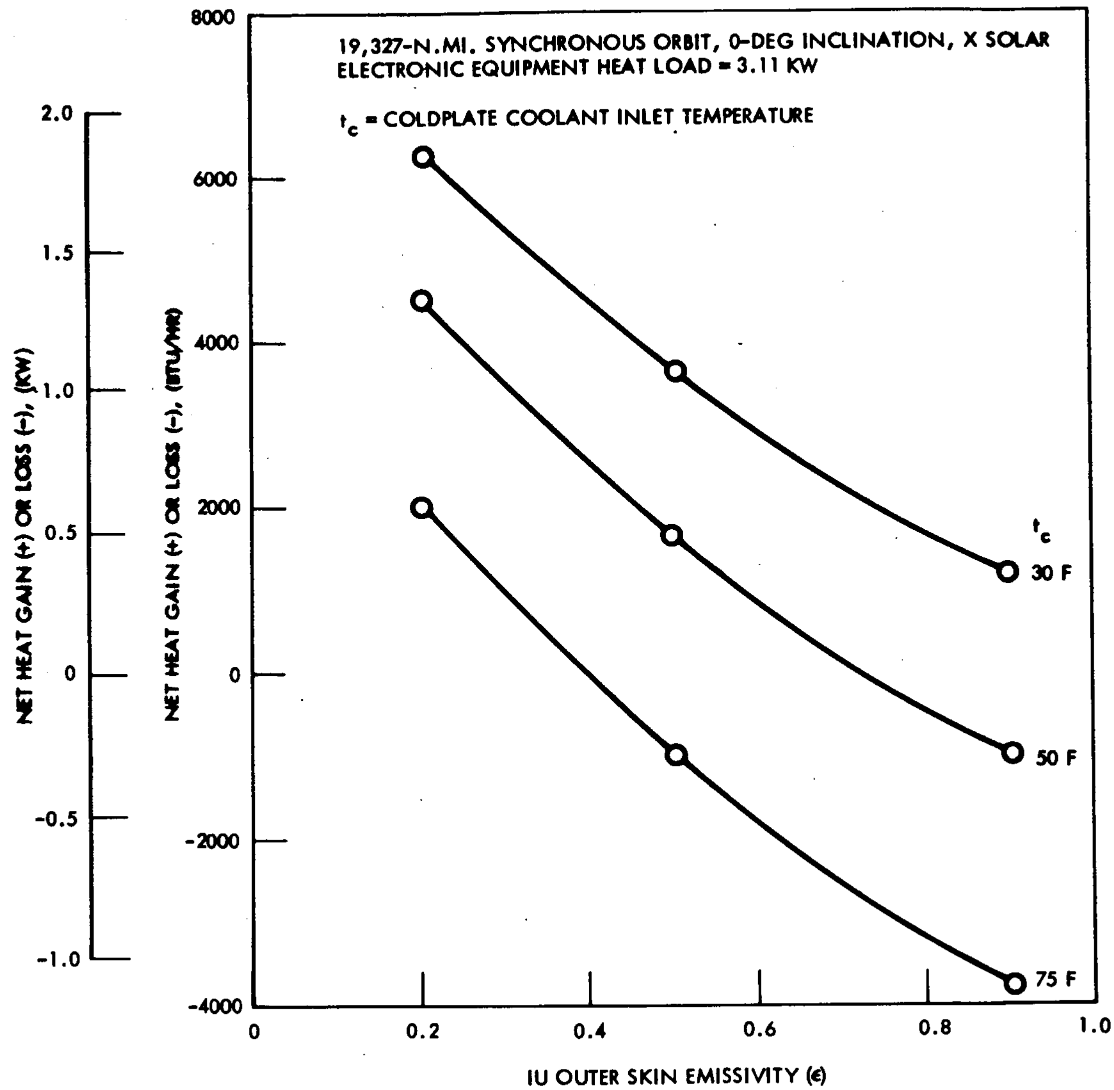

Figure A-34. Instrument Unit Net Heat Gain or Loss Versus Outer Skin Emissivity